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A. N. Ponomarev

THE YEARS OF THE SPACE ERA

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16. Abstract

In this book, the author describes further conquests of space. He describes new space vehicles of the Soyuz type, the orbiting spaceship Salyut, and the Apollo spacecraft used to land a man on the Moon, as well as their construction and rocket motors, and the cabin life-support system.

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A N N O T A T I O N

The author, a well-known aviation specialist, a Doctor of Technical Science, and a Colonel-General Engineer, has published a number of publications on aviation and space technology, including "Manned Spacecraft" and "Aviation on the Road to Space" in which he describes man's conquest of space in popular science terms. Space technology is developing vigorously, and long-lived vehicles of the Salyut type have been placed in orbit. Man has accomplished a landing on the lunar surface.

In his new book, Aleksandr Nikolayevich Ponomarev describes further conquests of space. He describes new space vehicles of the Soyuz type, the orbiting spaceship Salyut, and the Apollo spacecraft used to land a man on the Moon, as well as their construction and rocket motors, and the cabin life-support system.

The book is aimed at the reader interested in the development of space technology.

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THE YEARS OF THE SPACE ERA

A. N. Ponomarev

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The space research carried out during the last 15 to 20 years has opened up a new era in man's knowledge. Our country has expended a great deal of effort to create a high-power rocket technology, which in turn has enabled us to accomplish man's cherished dream, flight into space.

The first flight of a human in space, accomplished on April 12, 1961 by Yuri Gagarin, constituted a step into the unknown, breaking through the major psychological barrier which had to be overcome.

The subsequent program of Soviet space flight saw a steadily increasing complexity (from satellite spacecraft of the Vostok type to the orbiting space laboratory Salyut, which operated in conjunction with the Soyuz vehicle).

An outstanding achievement was the excursion into deep space by Aleksey Leonov. Group flights of space vehicles of the Soyuz type and transfer from one vehicle to another in deep space, and lengthy flights, and flights with vehicles docked together were accomplished during the Salyut flight with its heroic crew, G. T. Dobrovol'skiy, V. N. Volkov, and V. I. Patsayev.

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* Numbers in margin indicate pagination in original foreign text.

Soviet astronautics moved confidently ahead in its conquest of deep space, both in manned and in unmanned space vehicles. Man ventured beyond the limits of our planet. The possibility of future flights to other planets and to other worlds has expanded our ideas about the universe.

Space investigation has not only had a great effect on the development of science and technology, but has also found a wide practical application on Earth. Physical and geophysical investigations, atmospheric investigations and hydrological and meteorological observations will allow us to solve many problems in the very near future. The study of Earth resources, the obtaining of information about the surface of the Earth and even about the underlying rock, with the help of spectra of individual sections of the surface in the ultraviolet, infrared, and microwave ranges of the electromagnetic spectrum and the development /4
of exterrestrial astronomy and radio astronomy are presently extremely promising areas. The medical and biological experiments have had a great effect on the development of medical science.

On the whole, the development of astronautics has had a considerable influence on scientific and engineering progress and the development of many areas of applied science and technology. New types of metallic and nonmetallic structural materials have been created. New types of high-efficiency sources and electrical energy converters have been developed.

Astronautics has accelerated the solution of many problems in the area of automation and methods of transmission and processing of information, and has had a large effect on the development of microminiaturization in electronics, and the creation of small-size computers.

Space investigations have opened up new horizons for the progress of science, engineering, and industry.

The present book deals with the history of the development of astronautics, both Soviet and foreign. It makes use of data from the open Soviet and foreign literature. Chapter 8 is written entirely from material in the foreign literature.

CHAPTER 1

/5

BASIC DEFINITIONS. SPACE COMPLEX

The concept of space is rather a broad one. When speaking of space one usually has in mind the entire Universe. However, at present we are mainly considering flights of unmanned systems within the Solar System, and manned flight in the region of the orbits of Earth, Venus and Mars.

The basic force determining the motion of an artificial satellite or a vehicle in deep space is the gravitational attraction of the planet around which the given satellite or vehicle is revolving, or the gravitational force of the Sun (in the case of flight in deep space, far from a planet). The flight trajectory of an artificial Earth satellite or space vehicle in Earth orbit is determined mainly by the Earth's gravitational field. The trajectory can be determined with sufficient accuracy if one neglects the effect of the gravitational field of the Sun and the Moon. It is necessary to allow for these fields only if one desires results of very high accuracy. In an interplanetary flight of a space vehicle, the Earth's gravitational field has an appreciable effect only in the initial period when the vehicle is in the immediate vicinity of the Earth. At a certain stage of the flight the Earth's attraction can be neglected, since the basic influence on the flight will be due to the gravitational field of the Sun, until the vehicle is a certain distance from the destination planet.

Thus, depending on which gravitational field exerts the predominant influence on the space vehicle, flights in deep space are conventionally divided into flights around a given planet, when the forces due to other planets and the Sun are small and can be neglected, and flights between planets, when one cannot neglect the effect of the field of gravity of the Sun, and of the planets in particular sections. Flights around the Earth are called geocentric, flights around the Moon are called selenocentric, and flights between the planets are called interplanetary.

The motion of satellites and spacecraft (without applied thrust), like the motion of any heavenly body, is determined by the laws of celestial mechanics. The motion is described by Newton's law of universal gravitation.

Contemporary ideas concerning the structure of the Universe are based on the major discovery made by the brilliant Polish scientist, Nicolaus Copernicus (1473-1543), who disputed the ideas then prevalent of the Greek scientist Claudius Ptolemaios (Ptolemy) (140 A.D.) which asserted that the Earth is the center of the Universe and that all the heavenly bodies revolve around it. Copernicus put forward the hypothesis that the Sun is the central heavenly body, and that the planets move around it in their orbits. One of these planets is our Earth. It completes its path around the Sun in 365.25 days.

The German astronomer, Johannes Kepler (1571-1630), a stalwart supporter of Copernicus, established that each planet moved in an ellipse, of which one focus is at the Sun, in a plane passing through the center of the Sun. And the ratio of the squares of the time for the planet to revolve around the Sun is equal to the ratio of the cubes of their mean distances from the Sun.

The English naturalist and mathematician Isaac Newton (1643-1727) gave an exhaustive proof of the causes for the motion of the planets and formulated the law determining their motion. He observed that two bodies attract one another with a force depending, firstly, on the mass of each of them, and secondly, on the distance between them. This law, published by Newton, is called the Law of Universal Gravitation. The attractive force of the Earth appears to issue from its center. Each body lying on the surface of the Earth tends to fall as close as it can to the center, but is prevented from doing so by the Earth's surface. Thus a force arises with which the body presses on its support. This force is called the weight of the body, and is a consequence of the Earth's attraction. To prevent a body from falling in the Earth's gravitational field, one must overcome the attractive force by applying to the body a force which would act in the opposite direction to the force of gravity of our planet.

When a body moves in a circle a centrifugal force arises, acting in the direction opposite to the direction of gravity. When a body flies around the Earth with a velocity of 7.91 km/sec, the centrifugal force at all points of its orbit will be equal to the centripetal force. Neglecting drag forces, one can calculate that the body will neither move away from the Earth nor towards it, i.e., will fly in a circle, and will thus be an artificial satellite of the Earth. Such a satellite flight can continue indefinitely in theory (if outside the influence of the atmosphere).

Thus, the velocity of 7.91 km/sec. is an important one for describing motion of a body in space. It is called the circular, orbital, or first cosmic speed in the field of Earth gravity.

If a body directed towards space acquires a speed in excess of 7.91 km/sec, it will then move not in a circle, but in an ellipse. The larger the velocity acquired by the body, the more elongated is the ellipse. For a speed of 11.2 km/sec, the flight trajectory takes the form of a parabola in the coordinate system fixed in the Earth, and the body, overcoming the forces of Earth's gravity, will fly out along a closed curve into solar space. For this reason the speed of 11.2 km/sec is called the parabolic, or "escape" speed, or the second cosmic speed.

For a speed exceeding 11.2 km/sec, the trajectory will take the form of a hyperbola, which, like the parabola, is an open curve.

The so-called inertia of mass has a great effect on all types of motion. When its speed varies, the motion of a body experiences a well-known resistance, which requires an applied force to overcome it. This change in the velocity of motion in a specific time interval is called acceleration. The unit of acceleration is the change in velocity of a freely falling body at the surface of the Earth in a time of 1 second. It is equal to 9.81 m/sec^2, and is designated by the letter g.

An acceleration is conventionally described by a number which indicates the factor by which the acceleration exceeds the acceleration of gravity. Clearly, the larger the acceleration, the larger is the number.

An acceleration loading inevitably arises during an excursion into deep space. During the launch of a space vehicle, while the rocket engines are operating, the acceleration increases continuously.

We assume that an acceleration reaches 39.24 m/sec^2, i.e., the rocket gives an acceleration of 4 g (9.81 x 4). Therefore, the loading will also have a value of four units. This means that the "weight" of each member of the crew of the vehicle is increased by a factor of four. We put the word "weight" in quotes, since it has a relative meaning. Here it would be more correct to say that the force then acting on a person is increased by a factor of four.

The acceleration, and therefore, the loading can reach very large values. It is clear that a very large force will then act on a human being.

The size of a permissible transient loading on a human and its duration of action are found to be inversely related: the smaller the time of action of a gravity loading, the larger can the magnitude be. Therefore one cannot speak of a specific maximum loading which a human being can sustain. It all depends on the time of application. It has been shown experimentally that a person in a vertical position can sustain a transient loading of up to 8 g for 3 sec, and up to 5 g for 12-15 sec. For an instantaneous /9 action (less than 0.1 sec) the human being is capable of sustaining a loading of 20 and even more.

It follows from what has been said that one of the problems for designers of manned space vehicles is to arrange for flight conditions in which the loading is safe for humans.

One must remember that during flight in contemporary jet aircraft, large accelerations and correspondingly large loading arise in many cases. This loading is mainly centrifugal acceleration during a curved flight. These loadings may be generated in a ground laboratory by means of the so-called centrifuge. Experiments on a centrifuge and test flights have enabled us to establish

Academician S. P. Korolev and the first astronaut, Yu. A. Gagarin.

that a human being can withstand large loading if it is directed perpendicular to the longitudinal axis of his body, i.e., if the human being is lying down, since in this position the blood is not accumulated in any one part of his body. Some persons withstood loading up to 10 g and more for a period up to 30 sec in this position during tests. No disturbances were observed in these persons during the test.

TRAJECTORY ELEMENTS AND STAGES OF MOTION OF SPACE VEHICLES

The flight trajectory of the space vehicles (ballistic rocket, artificial Earth satellite, spacecraft, etc.) consists of the following sections: active section, passive section, and atmospheric entry (Figure 1).

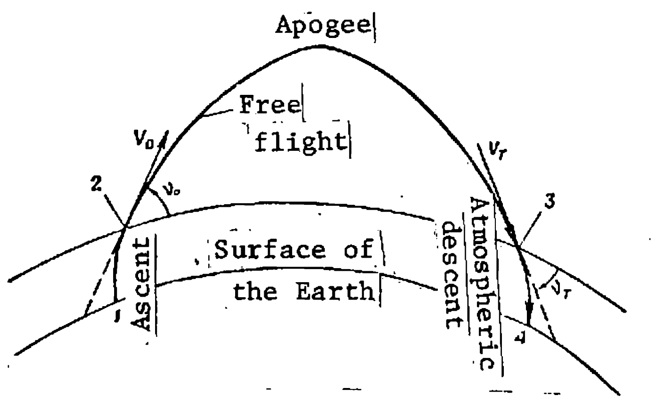

Figure 1. Flight trajectory of a ballistic rocket.

Active section (1-2). In this section the vehicle acquires the necessary speed for its subsequent flight. At launch of a space vehicle from Earth the speed of motion in the dense layers of the atmosphere is comparatively small. At the end of the active section, when the speed of motion is approaching cosmic, the flight takes place in the upper layers of the atmosphere. /10

The active section of flight for a space vehicle whose mission is a flight to the Moon, Mars, Venus, or other planets, can be divided into two parts. The vehicle is first injected into an orbit around the Earth, and then the engines are started which give the vehicle the second cosmic speed, and it transfers to a flight orbit directed towards the heavenly body.

Passive section (2-3). When a space vehicle has reached a given speed the engine is stopped. The active flight section then is ended. Further flight takes place under the action of the gravitational field of the given planet, one of its satellites, or the Sun. In addition, the space vehicle can be subject to an aerodynamic drag force in a planetary atmosphere, which can exert an appreciable effect on the flight velocity and the shape of the trajectory during a long flight.

Atmospheric entry section (3-4). For a return to Earth following a flight in space, a space vehicle is decelerated and enters the dense layers of the atmosphere. Deceleration is accomplished by a rocket motor. Space vehicles enter the dense layers of the atmosphere with near cosmic speeds. Then there is a sharp increase in the heat flux and of the aerodynamic forces acting on the vehicles.

MOTION OF A SPACE VEHICLE IN THE ACTIVE SECTION

The active section of the trajectory, in which the engines operate, communicates to the space vehicle the required values of velocity, height, and inclination of the trajectory to the horizontal. The nature of the motion of the vehicle in this section is determined by the flight program and the method of launch, since there can be not only ground launches, but also airborne launches.

By way of example of injection of an artificial satellite into orbit around the Earth, we consider the nature of its motion in the active trajectory section.

To inject an artificial satellite into a near-Earth orbit, the launch vehicle must give it the required height and accelerate it to a velocity at which it will continue to fly in a closed curve (orbit) around the Earth. The simplest orbit for an artificial Earth satellite is a circle, i.e., a curve in which the satellite moves with constant speed at a fixed distance from the center of the Earth. How does one determine the speed to which the satellite is accelerated so that it will fly in a circular orbit at different heights above the Earth's surface?

For a central gravitational field the force of gravity, in accordance with the Law of Universal Gravitation, is given by

$$G_r = mg_r = f\frac{Mm}{r^2} = \frac{Km}{r^2}, \tag{1}$$

where m - is the mass of the space vehicle;

g_r - is the acceleration due to gravity at distance r from the center of the Earth;

f - is the universal constant of gravitation; its value is the same for all bodies; $f = \frac{1}{3866^2}$ $cm^3/g\ sec \cdot sec^2$,

M - is the mass of the planet;

r - is the distance from the center of the Earth to the vehicle;

K=fM - is the gravitational parameter; for the Earth's gravity field K = 398,620 km^3/sec^2.

If the vehicle flies in a circular orbit, the force of attraction is equal to the centrifugal force for uniform motion at speed V in a circle of radius r, i.e.,

$$\frac{mV_{circ}^2}{r} = \frac{Km}{r^2}$$

Hence

$$V_{circ} = \sqrt{\frac{K}{r}}. \qquad (2)$$

This velocity, as has already been mentioned, is usually called the circular, orbital, or first cosmic speed.

For a circular orbit with a height of 100 km above the surface of the Earth, the orbital speed is 7.85 km/sec, and for a height of 300 km, it is 7.73 km/sec. Of course, this does not mean that it is simpler to inject a satellite at a height of 300 km than at a height of 100 km. The fact is that, as the height of the orbit increases, there is an increase in the velocity loss in overcoming the Earth's gravity force, and the growth of these losses as the height increases exceeds the decrease in the orbital velocity.

When considering the active section for injection of an artificial Earth satellite, one must take into account that the injection is accomplished from a rotating Earth. Therefore, the absolute final velocity of the satellite will be a combination of its velocity with respect to the Earth (relative velocity) and the velocity of the Earth's daily rotation. For constant absolute final velocity, the relative velocity of the satellite (the launch vehicle) can be less or more than the absolute final velocity, depending on the inclination of the orbit, i.e., depending on the geographic latitude of the launch site. The less the inclination of the orbit, the less is the required relative velocity, i.e., the greater is the use made of Earth rotation to communicate the required velocity to the satellite.

Thus, the greatest velocity which one can derive from Earth rotation, for zero inclination (the launch is performed in an easterly direction in the equatorial plane) is 465 m/sec. For large inclinations (more than 90°), one must even overcome the circular speed of rotation of the Earth. This means that the relative velocity must be greater than the absolute. /12

For injection of a satellite into orbit, an important matter is the program for varying the angle of inclination of the trajectory to the horizontal. At the first stage of flight, the trajectory of the launch vehicle is analogous to that of a ballistic rocket. It is a vertical launch section and has a comparatively steep initial section. Such a trajectory accomplishes the simplest launch and the fastest emergence of the satellite from the dense layers of the atmosphere before reaching large flight velocity. Subsequently, the control system achieves a given program for rotation of the rocket so that the normal forces continuously curve the trajectory in order to reach the prescribed inclination of the trajectory to the local horizon.

MOTION OF A SPACE VEHICLE BEYOND THE ATMOSPHERE

Coordinate systems. As has been mentioned, the motion of space vehicles, like that of all heavenly bodies within the Solar System, is determined mainly by the gravitational forces acting on them. The structure of the Solar System allows us, in many cases, to deal with free flight of a spacecraft in the gravitational field of one or other of the heavenly bodies. This enables us to describe the motion of planets, their satellites, and other heavenly bodies in coordinate systems whose origin is located at the centers of the gravitational fields. The corresponding coordinate systems are usually named for the heavenly body at the center of which they are located. For example, the heliocentric coordinate

system has its origin at the center of the Sun, the geocentric at the center of the Earth, the selenocentric at the center of the Moon, and so on.

Orbits of Artificial Earth Satellites. Depending on the velocity, height, and angle of inclination of a trajectory to the local horizon at the end of the active section, one can obtain different orbits for artificial Earth satellites. If the final velocity of the launch vehicle in the active section is circular, we obtain a circular orbit (Figure 2); if the final velocity of the launch vehicle is greater than circular for the given injection height, then the orbit will be elliptical. Then the point of greatest distance of the satellite from the Earth (the apogee) will always be at a greater height than the injection point. But if the final velocity is less than circular, then the point of least distance of the satellite from the Earth (the perigee) will always be less than the injection height.

For the same velocity and injection height, a change in the angle of inclination of the trajectory to the local horizon has a considerable effect on the perigee height of the orbit. The smallest perigee height is obtained for zero angle of inclination of the trajectory to the horizontal. An increase or decrease of this angle will reduce the perigee height and change /13 the satellite trajectory with respect to the Earth (Figure 2).

The time for one revolution of a satellite around the Earth (the orbital period) for a circular orbit of radius r is:

$$T_{circ} = \frac{2\pi r}{V_{circ}} = \frac{2\pi}{\sqrt{K}} \cdot r^{3/2}. \qquad (3)$$

The time of revolution of an Earth satellite in flight in a circular orbit at height 300 km, according to Equation (3), is about 90 min. As the height increases, the period increases very rapidly. It is easy to find the height at which the satellite period of revolution is equal to the period of daily rotation of the Earth: T_{circ} = 23 hr, 56 min, 4 sec. This height is equal to 35,830 km. A satellite launched in the equatorial plane in an easterly direction to this height remains above the same point of the Earth's surface. Such a satellite is called stationary, and its orbit is called a stationary orbit.

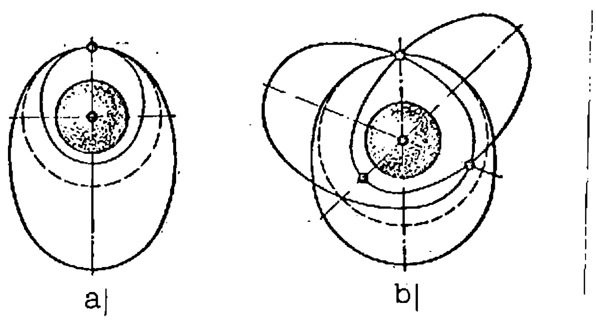

Figure 2. Change of satellite orbit.
a- dependent on the speed of injection; b- dependent on the direction of the injection velocity vector.

If the orbit is not circular, but elliptical, then the period of revolution is

$$T_{sat} = \frac{2\pi}{\sqrt{K}} a^{3/2}, \tag{4}$$

where a is the major semi-axis of the ellipse; and $a = \frac{r_p + r_a}{2}$ (r_p is the perigee radius; r_a is the apogee radius).

We now calculate the period of rotation of the first Soviet artificial Earth satellite (Figure 3), injected into an elliptical orbit on October 4, 1957. Its perigee height was 228 km, and apogee height 947 km. This means that

$$a = \frac{2R_з + H_p + H_a}{2} = 6959 \text{ km};$$

$$T_{sat} = \frac{2\pi}{631} \cdot 6959^{3/2} = 5770 \text{ sec} = 96.2 \text{ min.}$$ /14

This result is the period of revolution of the first Soviet artificial Earth satellite at the beginning of its orbit.

The lifetime of satellites depends on the height of the orbit into which they are launched. The orbit of a satellite will be constant if it is located at a height where the aerodynamic drag forces are very small.

Artificial Earth satellites presently play an important part in solving many scientific and practical industrial problems. Satellites are used to collect scientific data concerning the Earth and the Universe, for purposes of communication, transmission of television programs, navigation, weather prediction, and so on.

Increasingly wide use will be made of manned satellites, which will perform a number of completely new functions, for example, the rescue of crews in space, the servicing and repair of different spacecraft. In addition, in the USA, satellites are used for military purposes: a number of reconnaissance satellites has been launched, as is mentioned in the foreign press, to photograph regions of the Earth and various objects of military importance.

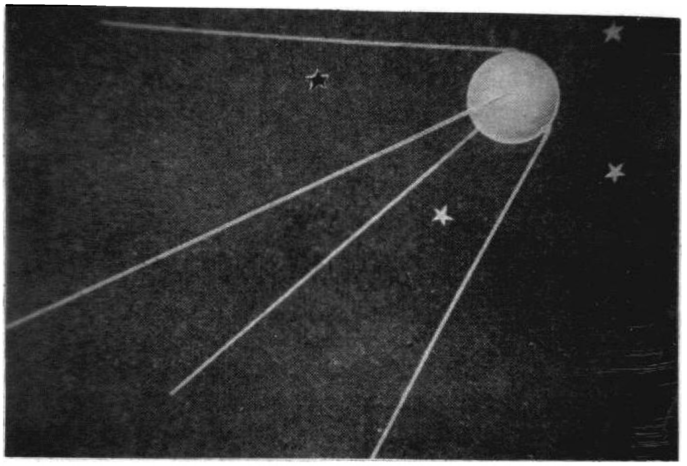

Figure 3. The first artificial Earth satellite.

The guidance of a satellite in the injection into orbit essentially reduces to guidance of the launch vehicle. After the satellite is injected into the design orbit, the requirements are as follows: stable motion in orbit and minimum departure from the orbit.

The guidance of satellites in orbit requires determination of a number of data to calculate the position. Following the appropriate observations which will allow the actual course and velocity of the spacecraft to be calculated, the required corrections in course and velocity are calculated for maneuvers which will fulfill the mission objectives. Navigational observations /15 are most effectively accomplished by means of ground tracking stations. However, by means of onboard instruments, which allow continuous observation of the ground position of the point geographically below the spacecraft, an astronaut in the satellite can determine the satellite orbital parameters, even when communication with Earth is lost.

Flights of space vehicles. If a spacecraft is given the velocity at which its kinetic energy $mV^2/2$ is equal to its potential energy $\frac{mK}{r^2}$, then the spacecraft escapes from the Earth's gravitational field. This speed, as has been mentioned, is called the parabolic speed, the "escape" speed, or the second cosmic speed. It is equal to 11.1859 km/sec (usually the value

11.2 km/sec is adopted).

Figure 4 shows the possible spacecraft flight region in height-speed coordinates. The figure also shows the flight region for ballistic rockets.

The spacecraft flight region is bounded below by the so-called aerodynamic barrier, and above by a line called the time barrier. The time barrier in this case is the flight duration (allowing for return to Earth) of not more than 10 years. The region is bounded on the left by the first cosmic speed, and on the right by a line where the acceleration exceeds 2 g. In addition, the flight region can be divided into sections in each of which it is most suitable to use a specific kind of rocket motor. This division is made in accordance with the velocity range which is optimum for the motor of the given type.

We now consider possible orbits for interplanetary flight and analyze some of the considerations entering into their choice.

The orbital planes of the planets either coincide with the orbital plane of the Earth, or deviate from it very little. Planets move along orbits close to circular. They are influenced by the gravitational acceleration due to the field of solar gravity (Figure 5). There are regions in space where the gravitational field of a planet dominates. Such a region is called the sphere of influence of a planet. The sphere of influence of a planet is the region of space in which one should consider the planet and not the Sun as the main center of attraction in analyzing the motion of a small body with the second cosmic speed relative to the planet.

The first manned interplanetary flight will undoubtedly use orbits of minimum energy, in which the force of solar gravitation will play the main part in the motion of the spacecraft. The flight of an interplanetary spacecraft will consist of a number of stages, each requiring an independent approach to the question of navigation, allowing for the influence of the gravitational field of the Sun, as well as of the departure and destination planets. /16

First stage, spacecraft launch. In this stage the spacecraft is launched from the Earth's surface (or the departure planet), is accelerated, and escapes from the Earth's field of attraction. At the end of the motor burn, the spacecraft has acquired a speed sufficient to escape from the Earth's sphere of gravitation. The Earth's gravitational field is the predominant influence on the spacecraft in this stage.

Second stage, flight in the intermediate section. In this stage, the spacecraft passes through the greater part of the distance between the planets, along a free flight trajectory or with a low thrust. The guidance problem in this case is to /17
determine the position of the spacecraft on the trajectory along which it must arrive in the vicinity of the destination planet in the given time. In the second stage one must calculate the original data for correction maneuvers and carry out these maneuvers. The gravitational field of the Sun has a predominant influence on the spacecraft.

Third state, flight in the approach to another planet and landing on it. In this stage the spacecraft transfers to the orbit along which it reaches the destination planet. The gravitational field of the planet has a predominant influence on the flight. The spacecraft enters the planetary atmosphere within the landing corridor, whose shape and size are determined by the characteristics of the planet and the spacecraft.

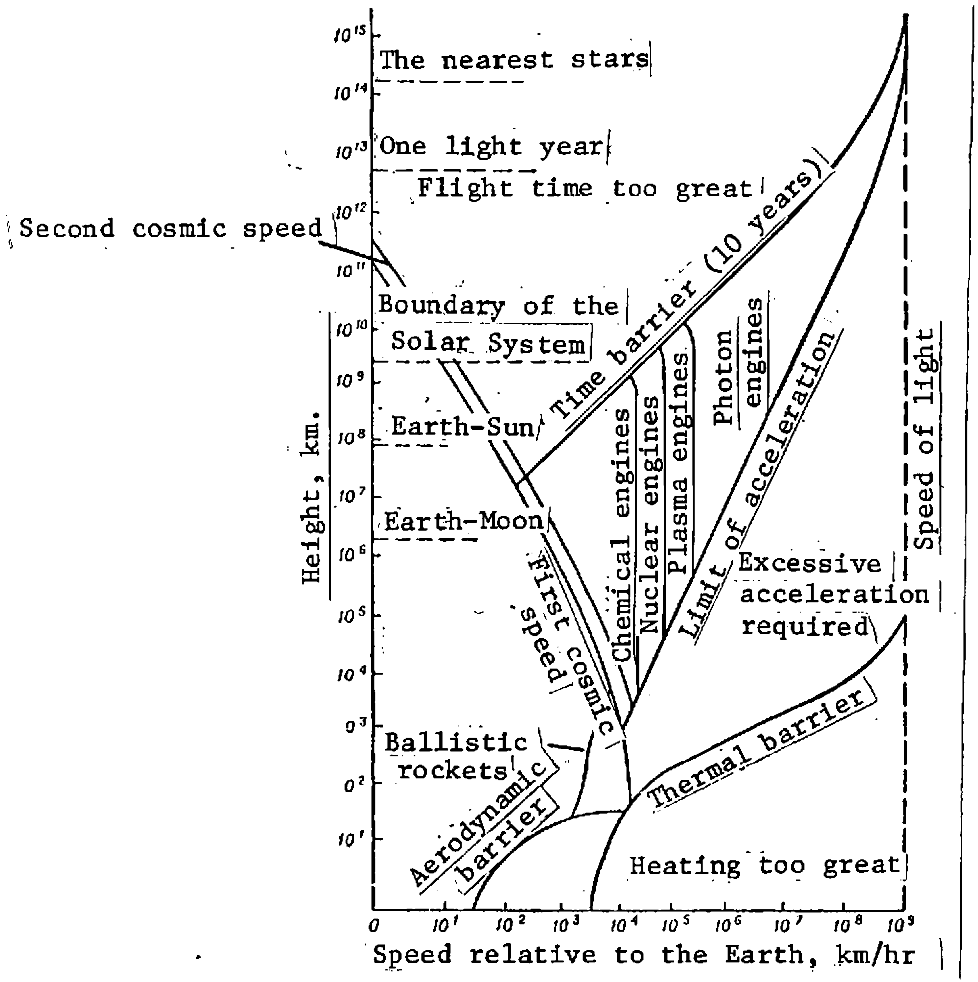

Figure 4. Regions of space-vehicle flight.

Fourth stage, return to Earth (or to the departure planet).

The duration of the first stage is determined by the radius of the Earth's sphere of attraction, which is about 925,000 km. The length of the second stage (heliocentric transfer trajectory to the nearest planet) is much greater, being hundreds of millions of kilometers. Thus, one can consider that the main part /18 of the middle section of the trajectory is located outside the Earth's sphere of attraction.

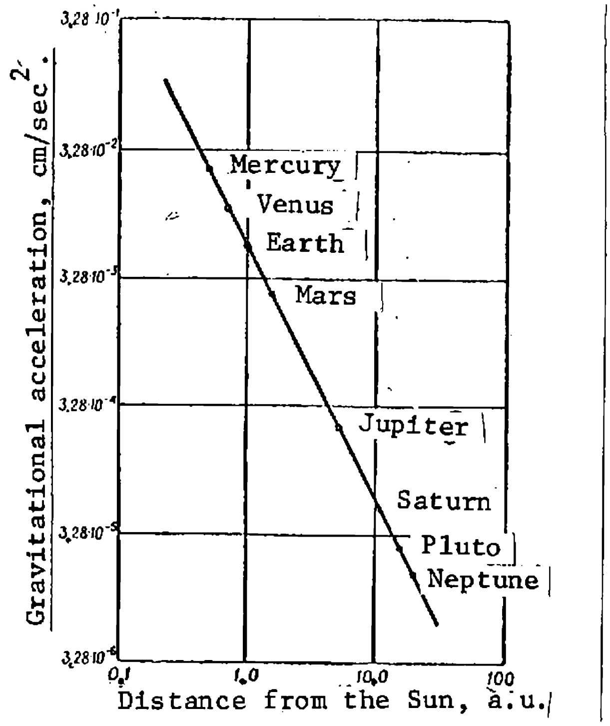

Figure 5. The gravitational field of the Sun.

If we have good knowledge of the initial coordinates and flight velocity of the spacecraft, as well as of the characteristics of the gravitational field in which the spacecraft moves, then, in theory, we can perform the spacecraft navigation without using external attitude references, celestial bodies, or special beacons. In this case the spacecraft must use an onboard inertial navigation system for attitude control.

The position and flight velocity of a spacecraft can be determined via measurement by astronomical methods, based on knowledge of the laws of motion of celestial bodies relative to the Sun. Stars or planets used as attitude references during measurements of this kind are in fact navigation methods of the same kind as attitude references and beacons used in sea or

air navigation on the Earth. The trajectory parameters of a spacecraft can also be measured by means of ground tracking stations. One can assume that in the future such stations will also be set up on heavenly bodies.

Spacecraft flights to the Moon have already become a reality. The Moon is the closest to the Earth of the heavenly bodies of the Solar System. It is a natural satellite of the Earth. The mean distance from the Earth to the Moon is 384,400 km. At perigee the distance to the Moon is 363,000 km, and at apogee it is 405,500 km. The mass of the Moon is 0.01 that of the Earth, the diameter is 0.27 that of the Earth, the volume is 0.02 that of the Earth, and the density is 0.61 of that of the Earth. The acceleration due to gravity on the lunar surface is 1.62 m/sec^2, i.e., it is less than that on Earth by a factor of about six. Therefore, the weight of every object on the Moon is less than on the Earth by a factor of six. The Moon always turns the same face towards the Earth. The reason is that it rotates about its own axis with the same period as that with which it rotates around the Earth. A flight to the Moon requires considerably less time than a flight to the more distant planets. Only a few days are required to fly to the Moon, while months and years are required for any other interplanetary flight.

The Earth and the Moon are often regarded as a single dynamic system. The common mass center of this system, or the so-called baricenter, is located on the line joining the centers of the two bodies, and is distant roughly 5000 km from the center of the Earth.

In contrast to the Earth, the Moon is almost devoid of atmosphere. Therefore, in a landing on the Moon it is not possible to dissipate the kinetic energy of the spacecraft by converting into heat, as can be done in motion in an atmosphere. Therefore,

the spacecraft must have a special system for deceleration and for absorbing the shock at the moment of touchdown.

<u>Docking of spacecraft</u>. An extremely important question in contemporary space flight is the rendezvous and docking of two spacecraft in space. On the one hand, one wishes to use minimum /19 energy in the launch of spacecraft and, on the other hand, there is a need to create orbiting scientific laboratories and to install in them supplies, spare crews, facilities for repair in space, and so on. We will take the term "docking" to mean the accomplishment of mechanical contact between two spacecraft. Docking is the final stage of rendezvous in orbit.

Development of a docking system was begun with the determination of general principles for its accomplishment. First, one must determine the compartment of a spacecraft most suitable for the docking fixture. This problem was solved by using the nose compartment for docking, since then one can exercise visual control of the docking process and observe it directly.

There is another problem in the choice of the kind of target, active or passive. An active spacecraft is capable of approaching the docking object by making corrections in its own orbit, using the radio beacon or flashing lights, and carries in itself the major part of the docking system; whereas a passive spacecraft cannot approach the docking object.

An investigation was also made of the possibility of repeated docking. A system designed for a single docking operation is rather simple. But the engineers have added to it a complexity associated with multiple usage. In order to evaluate the energy expended in docking and to choose the shock absorbent system, the docking process was investigated by simulation.

The docking is accomplished in stages: first there is a rendezvous of the two spacecraft in orbit, then a proximity maneuver, and finally the docking proper.

Rendezvous in orbit. To carry out a rendezvous of two spacecraft, the sequence of maneuvers required is as follows: a change in the orbit plane, transfer from one orbit to another, lying in the same plane, acceleration or deceleration in that orbit, and finally the approach.

The simplest method by which rendezvous of two spacecraft can be accomplished is the direct injection of an active spacecraft into an orbit lying in the flight path of the other spacecraft. Control of the launch vehicle which injects the active spacecraft for rendezvous is accomplished by the spacecraft from the launch point or by a system of companion stations, located along the flight track. In order to compensate for errors which can arise during the operation of the launch vehicle control system, it is customary to provide for maneuvers of the spacecraft in the final section of a trajectory prior to rendezvous.

The flight trajectory of the launch vehicle injecting an active spacecraft into orbit consists of several sections: injection, correction, search and target identification, and finally, homing guidance. The last section of active guidance results in rendezvous of the satellites. The passive spacecraft is located in a known orbit, and is continuously tracked, /20
i.e., its orbital elements are determined. These elements are fed into the guidance memory system of the active spacecraft. The time of launch of this spacecraft is determined with great accuracy with respect to the moment of passage of the spacecraft above it. The active spacecraft is launched in the same plane as the passive (reference) spacecraft, which ensures that there will be a rendezvous with the passive vehicle following a

correction maneuver. The scheme for rendezvous of the two vehicles in orbit by this method is shown in Figure 6.

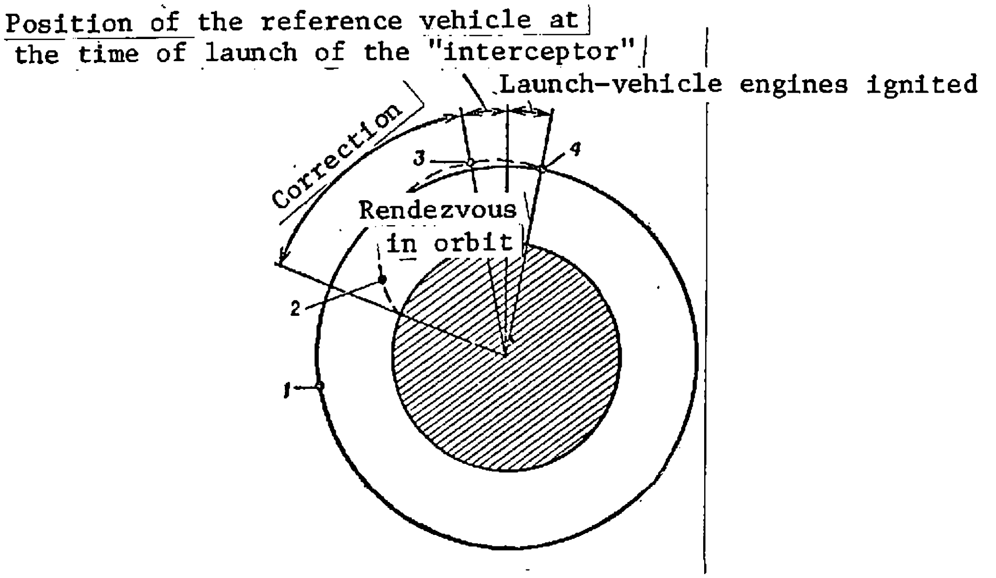

Figure 6. Scheme for rendezvous of vehicles in orbit.
1- search and target identification; 2- homing; 3- injection into orbit; 4- transfer ellipse.

A more general case is rendezvous of vehicles in orbit when the active vehicle is injected into an orbit which does not lie in the same plane as the orbit of the passive spacecraft. This requires maneuvers to change the orbital plane and maneuvers within the orbital plane.

We assume that the passive vehicle is located in a given orbit and that the active vehicle must be launched from the same launch point as the passive one was. As a result of Earth rotation the orbit of the passive vehicle projects on the surface of the Earth as an open curve (Figure 7). The launch point will lie in the orbital plane of the passive vehicle once every 12 hours. The only exception is the case when the spacecraft is launched into orbit with zero angle of inclination of the orbit, i.e., from

a launch point located on the Equator. Then the reference vehicle will have the latitude of the launch point twice in each revolution of the vehicle. If the angle of inclination of the orbit is equal to the latitude of the launch point, the track of the orbit on the Earth's surface will oscillate in the vicinity of the launch point latitude.

The approach maneuver for space vehicles differs from other maneuvers as regards the small relative velocities and distances. The initial conditions for the approach depend on the accuracy of operation of the spacecraft navigation and guidance systems in the final section. Naturally, if the speed of approach to the moment of contact is excessively large, then there is danger of damaging the docking vehicles because of the shock upon contact. Experience shows that a propulsion unit used for the approach maneuver must be one furnishing small accelerations. /21

The control of rendezvous of a manned vehicle and a passive one (laboratory) in the stage of close approach (relative range of the order of 15-30 km) and docking can be given to the astronaut. The reason is the exceptional ability of the human being to see three-dimensionally and to estimate the situation better than a machine when there are flight conditions not provided for in the program. The astronaut can make a flexible decision on the basis of inference, if he has a clear understanding of the situation that has arisen, in conjunction with prior knowledge and experience. However, the decision to have the astronaut take part in the control of the flight requires him to be free from carrying out various operations, thereby lowering the efficiency and reducing his ability to correctly perceive the surrounding environment and instrument readings. Therefore, for manual control, one must use methods of approach which ensure maximum reliability and can easily be accomplished by the astronaut.

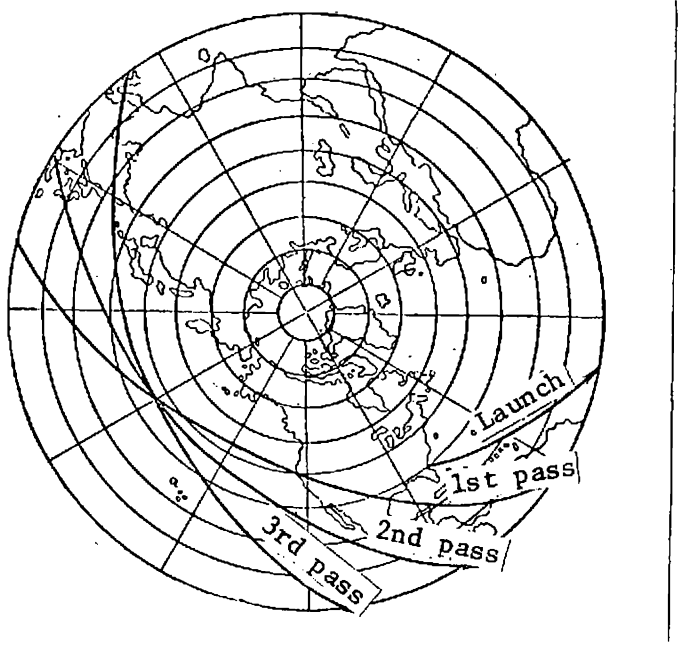

Figure 7. Polar diagram of the track of orbits (T = 90 min).

With reference to spacecraft this requires that the instrument system used for control should be simple, and the amount of fuel expended in performing the approach maneuvers should be minimal.

Physically the essence of the approach maneuver is that a change in the relative range is accomplished by deceleration, the result being that the relative velocity is reduced to an /22 acceptable final value. The problem of simultaneous control of range and relative velocity is the problem of controlling the magnitude and direction of the spacecraft acceleration.

Docking. It should be pointed out that the vehicles being docked in fact lie in different orbits. Their relative motion is affected by their having different values of gravitational attraction, but this effect is so small that it is usually neglected. In carrying out docking, the required directions of a

thrust may be different, and in general one must be able to create positive and negative thrust along all three of the body-fixed axes. To do this the active vehicle should carry a large amount of fuel and have an attitude control system capable of rapidly accelerating it in a given direction and accomplishing exact parallel displacement of the active vehicle relative to the passive one.

The control forces and moments during docking can be generated by electromagnets, in addition to rocket motors. Analysis shows that electromagnets with appropriate characteristics, mounted on the objects being docked, can generate magnetic fields that are sufficiently strong and capable of attracting the vehicles to one another and matching up their axes.

If a spacecraft uses an automatic homing system to accomplish docking, an onboard computer processes the control signals going to the control and attitude systems. An important factor in the choice of method of guidance is the determination of possible docking mechanism. If the mechanism allows change in the approach velocity over a wide range and does not require the approach to be rapidly executed, then it makes sense beforehand to establish an initial relative approach velocity and then perform the docking. If the time for the approach is in no way limited, and the final velocity should be maintained quite exactly, it is then desirable to establish the necessary final velocity at the very beginning and keep it constant.

Docking mechanisms have a special importance for docking of vehicles in space. They must reduce the difference in the velocities of the vehicles to zero; dissipate the kinetic energy; ensure a mechanical connection between the vehicles after contact; create repulsive forces which would allow these vehicles, if

necessary, to be rapidly separated; transmit electrical signals; pump fuel; transfer stores; and finally, transfer astronauts from vehicle to vehicle.

We shall examine one of these mechanisms for a lunar space vehicle and the sequence of its operation. The docking system was designed to connect the command module to the lunar excursion module and also to disconnect them. Docking was achieved by bringing the command module close enough to the lunar module to allow the tip of an extensible rod on the command module to drop into the socket of a receiving cone on the lunar module. /23

The main elements of the docking system (Figure 8) are the extensible rod, the receiving cone, and the docking ring. The command module and the lunar module are both equipped with sealed ports and manholes via which the astronauts transfer from one module to the other. The extensible rod unit consists of the rod itself, rocking levers, shock absorber pads, the rod latch, the preliminary capture latch, a ratchet mechanism, the extensible rod element and joints. The extensible rod takes the form of two aluminum cylinders (outer and inner) and is attached at three points to the docking ring by means of a support structure which can be folded and unfolded from either side, from the lunar module or from the command module.

The conical receptacle consists of a conical funnel whose internal surface faces the command module. The funnel is of honeycomb construction and made of aluminum. The docking ring is made of aluminum and is bolted to the wall of the command module manhole directly ahead of the upper hatch. Angular grooves contain the sealing rings and a pyrotechnic charge.

When the tip of the rod comes into contact with the conical receptacle, the tip slides along the conical surface and drops

into the socket, whereupon the preliminary capture latches operate, preventing the tip from withdrawing from the socket. The pilot of the command module switches on the rod extension unit, which draws the command module towards the lunar module. At final contact the 12 main latches operate, located on the junction ring of the command module, which ensure a rigid, sealed union. After the space vehicle departs into its selenocentric orbit (and /24 along a flight trajectory towards the Moon) the manhole between the two modules is opened, the hatch covers and the junction mechanism are removed from the manhole, and the two astronauts transfer to the lunar module. The astronaut remaining in the command module then transfers the conical receptable to the lunar module; the astronauts of the lunar module have placed this receptacle in the manhole. The pilot of the command module at this time operates the mechanism of the extensible rod, separating the joints, and withdraws the cables into his module, manually opens and lifts all 12 of the latches and fastens his hatch cover. The modules separate when the remote latch functions in the extensible rod mechanism. After the flight stage returning from the lunar surface with the main unit has docked, the pilot of the command module equates the pressure between the modules by means of a special valve and opens the hatch of the command module.

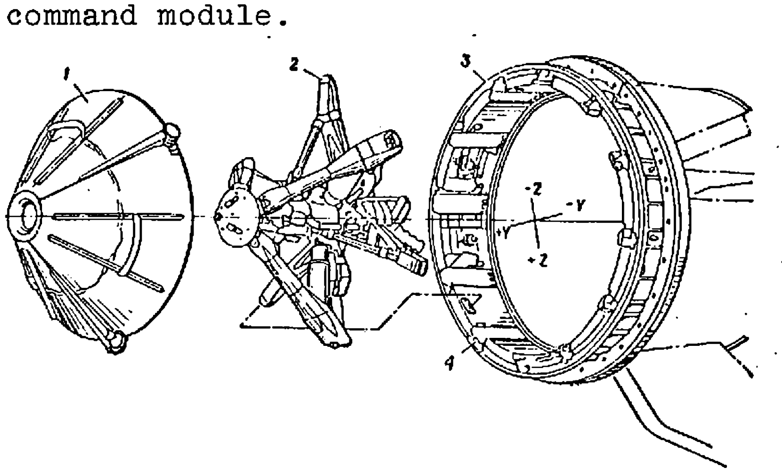

Figure 8. Basic elements of the docking system:
1- the conical receptacle unit; 2- the extensible rod unit; 3- the docking ring; 4- the main latch.

After checking that the main latches on the docking ring are in good order, the pilot of the command module withdraws all the docking mechanism from the manhole and stores it, together with the hatch cover, inside the module. Then he opens the valve in the cover of the docking hatch of the flight stage, for final equalization of the pressure. The cover of the flight stage can be opened from either side and swings on hinges inside the flight stage. The astronauts transfer to the command module. The covers of the two hatches are closed by the pilot and preparation proceeds for separation of the flight stage. The separation is accomplished by initiating the firing circuit, located around the docking ring. The docking ring separates as a whole from the command module, but remains attached to the flight stage.

Atmospheric entry

The culmination of a space flight along an interplanetary trajectory or in an artificial satellite orbit is a safe landing of the spacecraft on the surface of the planet. For planets possessing an atmosphere, this problem reduces to solving the problem of aerodynamic heating, of deceleration loading, of control of the time to reach the planet and to seek a landing site. In the absence of an atmosphere there remain only the problems of loading, control of the time to reach the planet, and the search for a landing site.

A space vehicle approaching a planetary atmosphere from deep space or decaying from an artificial satellite orbit possesses a large excess energy. This energy consists of kinetic energy due to the vehicle speed, and potential energy due to its position relative to the surface of the planet. Just as for any body entering an atmosphere with hypersonic speed, there is a strong shock wave ahead of the vehicle, at the nose section. The density and

temperature of a gas in the shock layer is increased sharply. As the dynamic loading of the spacecraft increases in the denser layers of the atmosphere, the vehicle is heated more and more /25 strongly, and its velocity decreases continuously as a result of aerodynamic braking. The kinetic energy of the vehicle is transformed into heat. If all the heat available were to go to heating of the vehicle, it would be enough to completely vaporize the vehicle structure. However, in actual fact, for example, in the fall of meteorites to Earth, a considerable part of the heat is given to the surrounding space by the shock waves. The transfer of heat by shock waves is a result of the interaction of the molecules of the gas surrounding the flying body. The layer of gas in which the particles interact is compressed to high pressure and heated to high temperature, and is bounded in front by the shock wave front. The shock wave extends far into the atmosphere on all sides from the vehicle and leaves a broad wake, formed by heated gas. The wake contains the main part of the heat liberated in the atmospheric entry of the vehicle. The heat flux reaching the vehicle surface comes through from the layer of compressed air mainly as a result of friction.

The heat passing out into the atmosphere is directly proportional to the strength of the shock wave: the stronger the wave, the less is the amount of heat transmitted to the vehicle as a result of friction. The strongest shock waves arise when the bow (or nose) of the body is blunted. Therefore, vehicles intended for atmospheric entry are usually given a blunted streamlined shape, and not elongated, as is the classical case of aerodynamics at subsonic and supersonic speeds (Figure 9).

Several methods are used at present to shield the vehicle structure from the flux of heat formed by the aerodynamic heating.

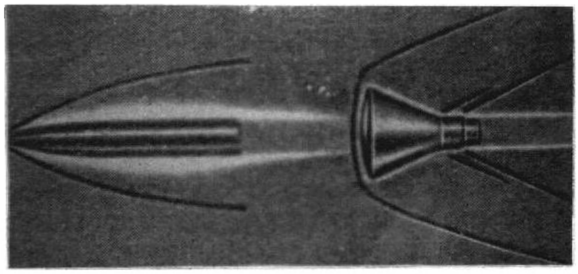

Figure 9. Picture of aerodynamic heating for bodies of different shapes, as a function of the shock wave strength (the dark lines)

If the atmospheric entry is ballistic or the entry occurs at large angle, the vehicle reaches the lower, denser layers of the atmosphere in a short time interval. During this time the vehicle is decelerated. Therefore, the heat flux, or the amount of heat reaching the vehicle in unit time, will be very great. /26
In this event, it is desirable to use a vehicle with a severely blunted nose section and a rather thick heat shield, capable of sufficient heat absorption. The thickness of the shield layer to absorb the heat is chosen so that the temperature of the rear face should remain below a value allowed for the material in question.

A greater time interval is necessary for a shallow atmospheric entry using lift from the space vehicle. The deceleration is accomplished mainly at very great height. Since the atmospheric density is small at these heights, the heat flux will also be small. It may happen that in the final analysis it is comparable to the heat flux radiated by the vehicle surface. In this case one can use the radiative heat dissipation technique that implies radiation cooling of a vehicle surface clad with a thin metallic sheath.

However, the simplest solution to the aerodynamic heating problem during atmospheric entry is now to use a heat shield made of insulating layers of fiberglass and other materials of that kind. The result of intense heating is that the outer layer of the heat shield melts and evaporates. The evaporating material absorbs a great deal of heat and thus reduces the transmission of heat from the shock wave to the vehicle.

Besides the problem of aerodynamic heating, there is also the problem of loading, which in some cases may be more difficult to resolve. Aerodynamic lift is used to reduce the loading during deceleration to values which human beings can withstand. This serves to reduce the vertical speed of descent and elongates the path of the vehicle towards the planet, therefore leading to a reduction in the loading. The lift is sometimes used even when the allowable loading is high, in order to reduce the heat flux, and also to control the time to arrive at the planet and the location of the landing point. A basic characteristic of a vehicle which possesses lift is the aerodynamic ratio of lift to drag, i.e., the ratio of the lift to the frontal drag. Space vehicles with a lift-to-drag ratio not exceeding 2 can accomplish a landing at any point of the surface within a range of thousands of kilometers in the longitudinal and lateral directions relative to the initial trajectory and the point of atmospheric entry.

Figure 10 shows the maneuver possibilities of a spacecraft with lift. The dotted lines show possible landing regions corresponding to different values of the lift-to-drag ratio (given on the curves). Each curve bounds a possible vehicle landing region with a constant lift-to-drag ratio. The regions between the curves correspond to possible locations of the landing point of spacecraft with a variable value of lift-to-drag ratio. The solid line shows a trajectory for a typical maneuver of a /27
spacecraft with a lift-to-drag ratio of 1.5.

Prior to entry into the planetary atmosphere, the motion of the spacecraft in the passive section of the trajectory is subject to the laws of celestial mechanics. This means that the vehicle moves under the influence of only inertial and gravitational forces. During the atmospheric entry, aerodynamic forces begin to act on the vehicle. An aerodynamic drag force acts in the direction opposite to the vehicle velocity. An aerodynamic lift force acts perpendicular to the vehicle motion. The gravitational force is always directed towards the center of the planet (Figure 11), and a centrifugal force acts in the opposite direction.

The dynamics of motion of a vehicle in the atmospheric entry section are determined by its own inertia and the resultant of the above-mentioned forces. The drag force reduces the vehicle velocity, while the centrifugal and lift forces give it acceleration in a direction perpendicular to its motion. The aerodynamic forces, like the forces due to acceleration, vary in direct proportion to the atmospheric density and the square of the vehicle speed. As the vehicle approaches the planet, it enters layers of the atmosphere with very low density. During subsequent penetration into the atmosphere, the density rapidly increases and because of the frontal drag, the vehicle speed begins to fall. Thus, the loading is proportional to the product of two quanti- /28 ties, one of which is increasing and the other decreasing. At some point of the trajectory the decrease in vehicle velocity begins to predominate over the increase in air density. The result is that the loading reaches a maximum value and subsequently begins to decrease.

It should be mentioned that during the return from deep space (in contrast with decay of an artificial satellite from orbit) a serious problem is the accuracy of control that would allow a given entry program to be accomplished, in particular to accomplish

descent in the most favorable trajectory and avoid excessively high loading and aerodynamic heating. A flight in a geocentric orbit does not require high guidance accuracy during atmospheric entry, since an entry that is too steep is easily corrected by short-term application of thrust, and for an entry that is too shallow, one can again apply a decelerating pulse. But during atmospheric entry at speed exceeding the first cosmic speed, guidance errors become very dangerous, since then an excessively steep entry can lead to the generation of loading which is intolerable to the astronauts or to disintegration of the vehicle during descent, while if the entry is too shallow, the result can be a skipout into space of the vehicle, which might not return.

Landing of space vehicles. A landing can be planned for the surface of water (as is the practice in the USA), but the possibility must always be envisioned of a landing on dry land (for example, during an emergency following launch). At the present time, landing of returning space vehicles on Earth, as well as landing of entry vehicles on Venus and Mars are usually accomplished by means of parachutes, which are relatively simple, reliable and accurate devices (Figure 12). But since ordinary parachutes have essentially a vertical and uncontrolled descent, they will possibly be used less and less. Besides an ordinary parachute, one might use a guided parachute which would allow the landing point to be chosen. Figure 13 shows a guided parachute which has been tested in the USA. The special feature is the flap. This is a part of the cupola, located at the rim and capable of being deflected upwards. Such a flap can offer only a limited possible choice of gliding trajectory.

In addition, for landing in an assigned area, one can use a planing parachute or create a device with a certain amount of aerodynamic lift.

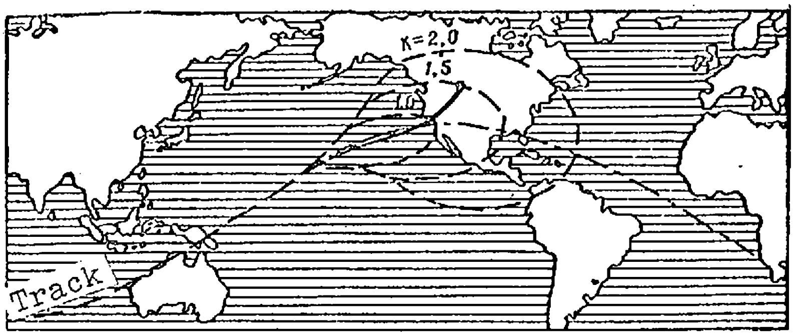

Figure 10. Maneuvering possibilities for a spacecraft capable of aerodynamic lift.

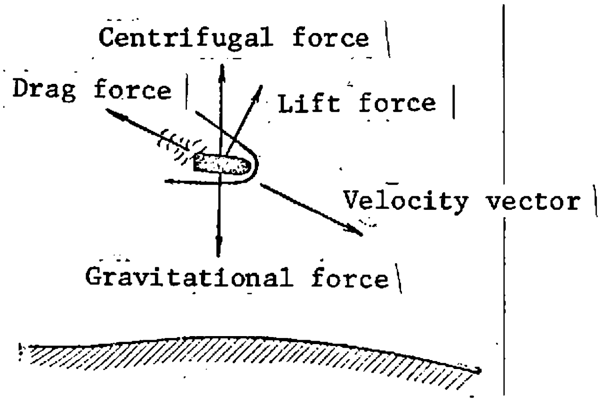

Figure 11. Forces acting on a vehicle during atmospheric entry.

THE CABIN OF A SPACE VEHICLE

During design of a space vehicle attention is given to the question of locating the control devices, instruments and equipment necessary for the crew to accomplish their assigned tasks, and to other matters (view from the cabin, emergency rescue, egress from the cabin, etc.). The ideas entering into the design of a space vehicle cabin are very similar to those entering into the design of an aircraft cabin. /29

The design of a space vehicle cabin is determined mainly by the task assigned to the astronauts. To relieve the pilot from control during cruising flight, autopilots have been developed whose gyroscopic section supplies signals to the control system to maintain a given spatial attitude of the vehicle. A similar

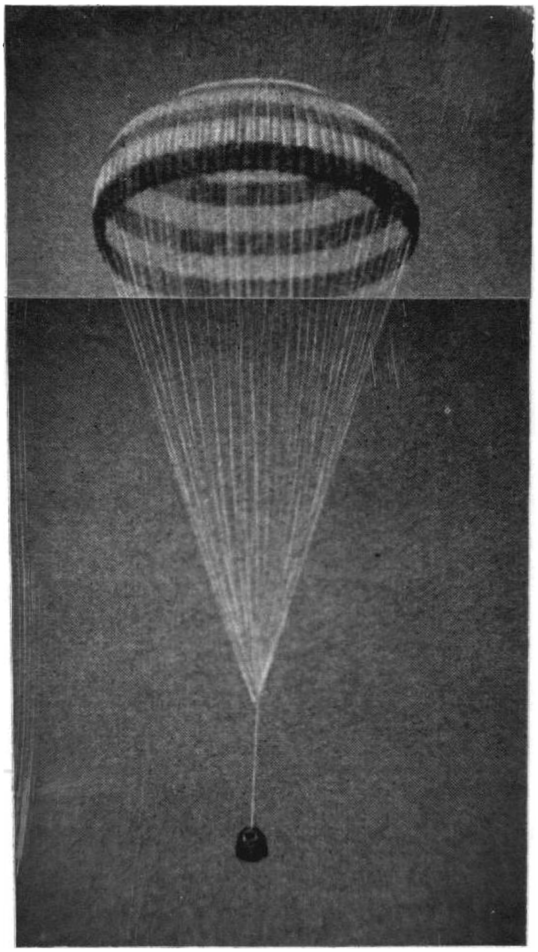

Figure 12. Descent of a spacecraft on a parachute.

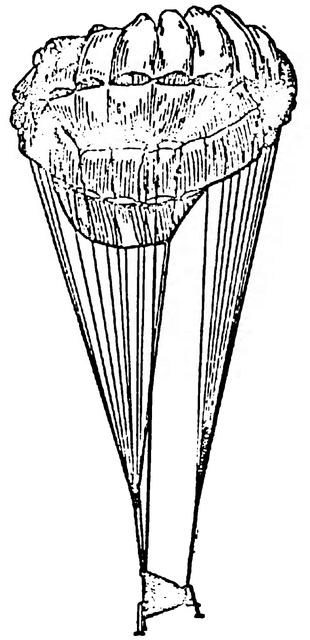

Figure 13. Descent of a spacecraft on a controllable parachute.

/30

system to relieve the astronaut has also been used on manned spacecraft. Besides the automatic attitude control, the space vehicle is equipped to ease manual control by an automatic stabilization of the vehicle. At large flight speeds in the atmosphere, the aerodynamic forces excite high-frequency oscillations of the vehicle. When the frequency of these oscillations approaches 1 hertz, the pilot's reaction breaks down. Damping of aircraft oscillations is performed by automatic systems which

generate control signals which will oppose the angular motion of the vehicle. This kind of damping of oscillations is also useful for control of a spacecraft during flight in deep space, where the absence of aerodynamic forces creates a rotation of the spacecraft with constant angular rate under the influence of an angular perturbation. Therefore, control devices located in the space vehicle cabin must enable the pilot to choose and carry out different combinations of modes embodying manual and automatic control of the flight.

The construction of many parts of a space vehicle cabin depends on the methods chosen for emergency rescue and for escape of the astronauts from the vehicle. An emergency rescue system on the launch pad must eject the spacecraft from the defective launch vehicle. The system must retain its ability to function up to the moment when the malfunction of the launch vehicle begins to present a hazard for the crew. Rescue of the crew can be accomplished by separating the entire space vehicle from the launch vehicle by means of a special rocket motor or by using an ejection seat for each crew member. The construction of the emergency rescue system must ensure the safety of the crew for any type of hazard on the launch vehicle, and should also provide means for absorption of shock. If ejection seats are used, large fast-opening hatches must be included in the space vehicle design, and there should be no obstacles along the exit path of the ejection seats. These hatches are also required for excursion of the astronauts into space and for leaving the space vehicle after a landing.

Because the natural illumination varies very rapidly during motion of a space vehicle in orbit, the artificial cabin illumination system should provide for manual control of the brightness of the light in order to create the necessary contrast for observation of instrument readings. Several factors must be taken into

account in developing the cabin illumination system. In deep space there are no atmospheric particles scattering sunlight. Therefore, the sky background is completely dark and the contrast of illuminated objects and their component parts is much greater than under terrestrial conditions.

Direct penetration of solar rays through the windows causes very strong contrast and highlights which can adversely affect conditions for observing instrument readings. Strong contrast can also make viewing conditions difficult if the passive space vehicle is located, during rendezvous and docking in orbit, between the Sun and the active space vehicle carrying out a maneuver. Rapid and frequent transition from daylight to darkness when the vehicle enters the Earth's shadow requires rapid adaptation of the eye to the change or the use of filters on the windows in the daytime or spectacles with filters. Adaptation to darkness is required by the navigator before observing the stars, and also for the pilot during a landing on the Moon, if the landing is carried out on the dark side while the Sun illuminates the surface of the Earth. For this case, one must provide illumination of the cabin in the lunar landing module of a type which provides optimum contrast in the different sections of the descent, landing, and ascent.

In the ideal case, a very large part of the surface of a space vehicle would consist of windows. This would afford the astronaut visual control of some of the prior-to-launch operations would ensure attitude control in the active section of flight and would allow the astronaut to see the separation of the accelerating stages. Good viewing conditions are also required for rendezvous in orbit, docking, observation after firing of a retro-engine and during landing. In addition, by observing through windows one can achieve high accuracy in the space vehicle attitude. However, because of the limitations as regards weight,

and allowable temperatures and pressures, the number of windows and their size must be reduced to a minimum.

Control of rotation of the space vehicle relative to all three axes is accomplished using one hand by means of a light movement of the wrist. Because there is no need to move the arms and legs, as would occur on an aircraft, the influence of the astronaut on the control of the vehicle is reduced when there is deceleration during atmospheric entry. The manual control can be reduced to the action of a valve in the jet control system by means of a mechanical linkage, to supply an electrical signal directly to the winding of a solenoid valve or by supplying a signal to a solenoid valve to increase the damping of oscillations.

The thrust of the engines of the system for jet control of the forward motion cannot always be directed through the changing center of gravity of the space vehicle, and there must, therefore, be an interaction between the rotary and translatory motions. This makes it necessary to have simultaneous control of rotary and forward motion, which in turn requires two control arms, one for each hand.

The composition of the instrument panel of a space vehicle is analogous to that in an aircraft. The instrument panel contains instruments of two basic types: instruments showing the flight /32
conditions of the vehicle (angular speeds, angular position, speed of forward motion), and instruments showing the parameters of the vehicle systems (pressure, temperature, amount of fuel). Additional instruments may be necessary for monitoring the operation of fuel elements, onboard computers, navigation instruments, and of the launch vehicle.

A space vehicle is joined to the launch vehicle for a very small part of the time, but the joint flight in the active section

is most critical from the point of view of the safety of the crew members. The tendency to increase the role of the astronaut in controlling the vehicle during emergency rescue in the active flight section results in his instrument panel having a considerable number of instruments belonging to the control system of the launch vehicle and for sending commands relating to emergency rescue. The location of the main instrument panel immediately in front of the astronauts is dictated by the need for continuous monitoring of the most important parameters of the launch vehicle and the parameters associated with control and monitoring during atmospheric entry. Therefore, these instruments have a preferential location in the center of the instrument panel.

THE SPACE COMPLEX

At present the flight of a space vehicle is supported by a large complex of ground facilities. The launch into orbit uses a multistage rocket, called a launch vehicle. The launch pad and facilities for final assembly and pre-launch checking of the rocket and vehicle constitute the launch complex. The spacecraft (or space vehicle), the launch vehicle, the launch complex, and the flight complex constitute the space complex.

A manned spacecraft is constructed in order to send a crew into space to accomplish a specified mission. The structure of such a space vehicle is determined by the conditions of space flight and by the mission. The vehicle must be provided with an artificial, constantly renewed atmosphere, must have meteoritic and radiation shielding, and must be equipped to emit heat from the surface via radiation. A space vehicle is located in a practically inviscid medium and flies under its own inertia under the influence of the attraction of the planets of the Solar System.

To change the trajectory or the flight velocity, one must not only expend a certain amount of energy, but must also eject a certain amount of mass.

The geometric shape of a spacecraft is determined by the requirements for flight in space as well as in the atmosphere during injection into orbit and return to Earth. In order to reduce the loading on the launch vehicle, and to control the flight, the shape of a spacecraft must be symmetrical. During the atmospheric entry such a compact spacecraft shape allows reduction in the area of surface to be protected from the action of the large heat load, so that the mass required for this should be a minimum. The ratio of the spacecraft volume to its cross-sectional area should be a maximum, to reduce the radiative and meteorite danger and the probability of breakdown of the cabin sealing.

Ideally the atmosphere in the cabin of a space vehicle should be the same as on Earth at sea level. However, an atmosphere of pure oxygen is sometimes used, at a pressure below that of sea level, or a mixture of oxygen and an inert gas. In order to use the oxygen efficiently it is purified and reused. Water vapor and carbon dioxide must be eliminated, and the oxygen used again. The cabin of a space vehicle, as was mentioned earlier, is in many ways similar to that of a contemporary aircraft.

During flight the astronauts may wear pressurized suits. However, it is tedious to remain in pressurized suits during a long flight, and therefore these are donned only when needed.

In case of emergency, a spacecraft may land in a remote region of the land or ocean, and therefore the astronaut (like a pilot) is provided with emergency rations necessary for life support.

Rocket engines are used for maneuvers in space. They provide either small changes in the trajectory, or considerable changes of flight velocity. In the latter case, a significant part of the initial mass of the space vehicle must be set aside for fuel. In any case, maneuvers in space are necessary for successful accomplishment of the mission and for the safe return of the astronauts to Earth.

The simplest propulsion unit consists of solid fuel motors, and can be used during the descent from orbit. A more complex propulsion unit provides maneuvers at rendezvous of space vehicles in orbit, and also for changes in the vehicle orbit to achieve a landing at the chosen planet. In this case, the vehicle must be equipped with several motors in order to accomplish a landing of some of the crew on a planet and return all the crew members to Earth.

The source of electrical energy for equipment on a space vehicle can be: chemical batteries, solar cells, fuel elements, and heat engines operating with chemical or nuclear fuel. The choice of energy source is determined by the flight duration and by the loads. Usually AC and DC sources are carried.

The onboard systems for navigation, attitude control and guidance are planned in accordance with the flight mission.

During flight an onboard computer must process a large amount of information and accurately calculate the basic and and correction maneuvers, so that the vehicle can accomplish each section of the trajectory with minimum expenditure of fuel. During each maneuver, the inertial measurement unit must interact /34 with the computer to control the space vehicle, in order to change the direction of the velocity vector in exactly the required manner.

In spite of the fact that manned space vehicles are equipped with a unit to perform automatic navigation, the astronauts receive help from ground tracking stations which interact with coordinate computation centers, to increase the accuracy of the navigation computations carried out onboard the vehicles.

Manned space vehicles are equipped with several electronic systems providing two-way communication with Earth, and telemetry and external trajectory measurements. In addition, electronic systems are used to facilitate search for the vehicle following its descent to Earth. Radar equipment is carried on the space vehicle for docking in orbit or for a landing on a planet. In the event of the onboard navigation system malfunctioning, uninterrupted communication with the ground facilities is provided, in order for the mission to succeed and for return of the vehicle to Earth.

The internal space of a space vehicle. An increase in the volume of a space vehicle increases its mass. In order to increase the useful volume of the vehicle, one must make the basic structural elements stronger, and consequently, heavier. Space vehicles can be made smaller in volume with a decrease in mass. Effort is expended to locate the various systems inside the vehicle in a more compact manner in order to reduce the space occupied by them to a minimum. In addition, the position of the center of gravity of the vehicle must be held very accurately constant, and this creates additional difficulties in the location of the equipment. In addition, some elements of equipment can be located only in specific places or compartments. It is clear that the problem of equipment arrangement is made easier if the vehicle volume is increased or if the size of the equipment is reduced.

It is useful to provide for some margin in the internal volume of a vehicle in designing it, if its mass will allow this.

It not only eases the development of the structure, manufacture, checking, operation, and servicing of a space vehicle if it is designed with some excess volume, but it also increases the reliability, and additionally allows the vehicle to be used for other missions.

Models are used when deciding the arrangement and location of equipment, so that one can examine the different variations in the arrangement of equipment by a method of successive approximations.

Reliability. Manned spacecraft must be operationally reliable, since they carry the astronauts into an unknown environment, which is not easy to penetrate in order to render assistance. Therefore, reliability is the decisive requirement in /35
planning and developing a manned spacecraft. Each structural design can be accepted only after its effect on the reliability of the vehicle has been appraised and gives rise to no concern. The technology of the manufacture, the materials, the control, method of transportation, overall check and other areas of the development of a manned space vehicle must receive careful study, and all the initial proposals must meet the requirements of absolute reliability.

One can cite several ways to increase the reliability of systems:

1) Duplication of elements and systems which could fail, or provision onboard of spare items of different structural types, capable of performing the given functions;

2) Simplification or strengthening of systems, the objective being to reduce the number of elements subject to failure, or to eliminate them completely;

3) The use of the most suitable and reliable systems.

A substantial improvement in the characteristics of manned space vehicles can be achieved by eliminating unnecessary equipment. In the flights of the American Mercury satellites, the emergency rescue system motor was jettisoned immediately after separation of the first stage of the Atlas launch vehicle, since it was established that after this time emergency rescue would not be required since the vehicle is outside the dense layers of the atmosphere, and a failure of the rocket would not cause a large loading on the cabin. Jettisoning of the motor of the emergency rescue system reduces losses due to its mass. It is also sensible to jettison the spent retro-motors prior to entering the dense layers of the atmosphere during return of the vehicle and to retain only the descent capsule. This decreases the mass of the returning vehicle and correspondingly allows the mass of the vehicle and its heat shield as a whole to be reduced. The spacecraft Soyuz and Apollo jettisoned a very considerable mass before entering the dense layers of the atmosphere; in particular, they jettisoned elements of the thermal control system, electrical supply sources, the motors of the attitude control and maneuver systems, and also the retro units, the equipment module and the orbital module. This significantly increased the flexibility of the structure and indirectly achieved a large economy in mass, as well as improving the characteristics of the vehicle and increasing its reliability.

A spacecraft is made up from large isolated modules, this being a conventional approach which is advantageous when stages are separated and when parts of a spacecraft are jettisoned after they have functioned. In particular, the Apollo spacecraft, in its flight to the Moon, consisted of a crew module, an equipment module, and a lunar module. The crew module contained the astro- /36 nauts during injection into orbit, during the atmospheric re-entry, and during the major part of the flight. The equipment module housed the propulsion unit, the power supplies and the stock of oxygen for the life-support system.

The propulsion units of manned spacecraft have typically very high requirements regarding reliability of the motors, and differ from ordinary motors in that the increased reliability is achieved by simplifying the structure and using duplicated elements. In addition, in planning the propulsion units of spacecraft, larger margins of fuel stock are provided than for launch vehicles. The excess margin of fuel increases the probability of accomplishing the mission, since it ensures continuous operation of the motors in unforeseen circumstances and with navigational errors.

Launch vehicles used for the launch of manned spacecraft are not significantly different from ordinary launch vehicles and are modified ballistic rockets.

At the present time, multistage launch vehicles are used to reach a large velocity at the end of the burn section. Neglecting loss of velocity from gravitational forces and drag resistance, one can calculate that the final speed of a single stage rocket is equal to the natural logarithm of the mass factor (the ratio of the initial mass to the mass of the rocket at the end of the burn), multiplied by the specific impulse and the gravitational constant. Thus, the mass factor is the parameter which determines the flight

speed. In practice the mass factor is limited by the structural capabilities. It can be calculated that the speed of a single stage launch vehicle is limited to about 6000 m/sec, without using high-energy fuel, which can increase the speed by roughly one third.

In order to achieve high efficiency, the individual stages of a multistage launch vehicle are chosen so that each stage increases the speed by 4500-4600 m/sec. However, it should be noted that, to obtain such a speed increment during flight at low speed, a considerably greater fuel flow rate is required than for flight at high speed. The reason is that during flight at low speed the spacecraft has a large mass and greater fuel flow rate is required to obtain the same increment of speed. In fact, if all the stages have motors with the same specific thrust, then the same speed increment will be achieved for the same fraction of expended mass.

<u>The launch complex</u>, in addition to the launch area, which contains the launch pad, the service tower, and the bunker for the launch crew, also includes special facilities designed for final testing of systems, weight and balance measurement and other preparatory operations, test facilities for pyrotechnic tests of the various propulsion units and control motors, and assembly shops where the mating of individual stages of the launch vehicle and the mating of the spacecraft with the /37 assembled launch vehicle are tested.

<u>The control complex.</u> The flight control center, part of the support complex, is the main command point. The center personnel has at its disposal all information concerning the flight. They know the flight trajectory and the spacecraft position on each trajectory. The center computers predict the motion of the spacecraft and determine the required correction maneuvers. Special operators monitor the operation of the main

onboard systems. The physical condition of the astronauts is monitored by recording instruments. Practically all the information bearing on the flight is brought together and converted to the most convenient form, and comes to the operators who can render assistance to the astronauts at any moment of the flight.

A net of measuring stations is connected to the control center. These are sited so that the spaceship is within the range of view of one or several centers for the greater part of the flight. These stations are equipped with tracking antennas and means of two-way radio communication with a spacecraft, which are ordinarily the communication link between the spacecraft and the flight control center. Some of these stations have equipment enabling them to take control of the spacecraft at critical moments of the flight, to avoid depending on ground lines of communication with the flight control center.

A spacecraft can be landed in a region which was not envisioned in the mission design. More than half of the area of the Earth's surface is a possible landing region for a spacecraft in an emergency. However, in practice certain specific regions are set aside for an emergency landing. There are a number of search and rescue services, equipped with ships, aircraft and other ground and air transportation facilities, available in a region of normal or forced landing.

HUMAN PHYSIOLOGY IN THE SPACE ENVIRONMENT

For a human to exist in space conditions a certain pressure and composition of the surrounding gaseous medium are necessary.

Pressure. When a human is subjected to a low barometric pressure, a malfunction of specific organs and of the entire organism can take place and can lead to death. The cause is a

lack of oxygen and boiling of the fluid in the human body. The main cause of this phenomenon is not the reduced barometric pressure of the medium, but the reduced partial pressure of oxygen in the inhaled air. A continuous supply of oxygen to the /38 whole human body is necessary for the maintenance of life activity. Oxygen is supplied to the body as a result of circulation of the blood, which is saturated with oxygen via the lungs. The mechanism of breathing ensures contact between the oxygen of the air and the very fine blood-carrying vessels of the lungs, and also removes carbon dioxide gas, the end product in the oxidation of food, from the blood and the lungs. At normal barometric pressure the partial pressure of oxygen in the lungs and in the arterial blood is 100 torr, and the blood in the vessels of the lungs is sufficiently saturated with oxygen. At reduced barometric pressure, the degree of saturation of the blood with oxygen is reduced, since the partial pressure of oxygen in the inhaled air is reduced. Therefore, if a human being is subjected to the action of space vacuum, the partial pressure of oxygen in the inhaled air will be zero, the cells of the brain are deprived of oxygen within 5-10 sec, and the person loses consciousness. If the supply of oxygen to the body is not restored within 2-3 min, the brain cells experience irreversible damage and the person will die. The tissues of the body can live with a partial pressure of oxygen in the blood of 60 torr, but then certain physiological deviations arise, in particular, a reduction in the ability to resolve intellectual problems.

It has been established by physiological investigation that a barometric pressure of 198 torr is the minimum required for a person breathing pure oxygen.

As mentioned in the foreign literature, a pressure of 180-190 torr is maintained in space suits, and they are supplied

with pure oxygen. The astronauts spend a good deal more time in the cabin atmosphere than in space suits, and therefore the requirements for the cabin atmosphere are different.

It has been established that for lengthy time periods, one should not use an atmosphere of pure oxygen, since it has a harmful effect on humans. Even so, for periods up to 14 days, there is very little harmful effect of an atmosphere of pure oxygen. The addition of an inert gas to pure oxygen averts certain harmful consequences of breathing pure oxygen. There should not be any other gas in the cabin of a spacecraft or in a space suit, since if the concentration of such a gas is large enough, it is possible for the oxygen to be expelled, and its partial pressure reduced. In addition, the foreign gas might have a toxic effect on humans. At normal pressure for life on Earth, the carbon dioxide gas liberated by the lungs is rapidly absorbed in the surrounding air. If the ventilation of a spaceship or a cabin is insufficient, this kind of increase in the partial pressure of carbon dioxide can occur, and the human body cannot compensate for this.

With relevance to the oxygen needs and the liberation of carbon dioxide, one should remember that the main source of energy for mechanical, electrical and chemical operating processes taking place in the human body are the carbon-hydrogen bonds of the organic products of food. The energy of these bonds is liberated for the human organism as a result of complex oxidizing chemical reactions, for which oxygen is necessary in the final analysis. The products of these reactions are carbon dioxide /39
and water. The greater the physical load on the astronauts, the greater is their requirement for oxygen.

The ideal temperature inside a space vehicle and space suit is a temperature at which the astronauts do not experience any physical stress. Therefore, one must maintain normal thermal conditions for the astronaut's body; the protective properties of the human body against heat and cold are used only in emergency conditions. The mean temperature of the skin should be about 32°C. All the heat reaching the skin (both from the external medium, and as the result of metabolism in the human body) must be eliminated at the same rate at which it is produced. Removal of heat must be accomplished at the above mean skin temperature.

In order to maintain appropriate thermal conditions, the crew must be able to control the temperature of the cabin medium.

Food. Ordinarily the amount of food required by astronauts in flight should correspond roughly to their rations on the ground. The most reasonable method of combining good taste and low mass during flight is to use frozen and dehydrated food. In preparing this food during flight, one adds water which has been heated with heating elements.

Food in a frozen and dehydrated state is almost no different from its original form. In addition, it can be stored for a long time, protected from bacterial and chemical action, is simple to prepare, has a small volume, and does not require sterilization.

For a lengthy space flight, a human being should receive 600 g of food daily, consisting of 14-21 % by weight of fat, 14-20 % of protein, and 59-72 % of carbohydrate. This corresponds to a daily calorie intake (2800 kcal) that is quite sufficient for a human being in a normal flight in a space vehicle cabin. If the physical exertion increases, for example, when working in a space suit in space, this allowance must be increased.

The astronauts' rations must include food rich in protein, since this promotes the formation of nitrogen in the human body.

The human body loses water via imperceptible evaporation (this loss is small and constitutes 0.8 g/h), by exhaling moisture (loss on the average is about 1.35 kg daily); by perspiring, which depends on the surrounding temperature, and sometimes on psychological factors; via the urine (normal daily loss is about 1.35 kg), and finally via the feces,(the daily loss is about 1.35 kg). /40

Water enters the human body along with the food, and is also formed inside the body by chemical reactions. The body should maintain a constant amount of water. To do this, the amount of water entering the body with food and formed from chemical reactions should be equal to the total loss of water from the body.

Ionizing radiation. Under normal terrestrial conditions, all humans are subject to the action of small dosages of ionizing radiation coming from natural (secondary cosmic rays, or radioactive layers of rock) or artificial sources (X-rays). In space, the intensity of ionizing radiation is much greater than on the Earth's surface. The hazardous types of radiation are primary cosmic rays, protons and electrons from the radiation belts, gamma rays and neutrons from nuclear reactions in space and the flux of protons formed during solar flares.

Under space flight conditions, the radiation dosage should not exceed a specific value, since the normal activity of the human body will otherwise be perturbed. This disturbance may appear immediately, or after a short time interval, or after several years. The maximum radiation dosage acting on the unprotected skin of an astronaut during a very strong solar flare is

1500 rad and is hazardous for humans. Therefore, during space flights which may last for a considerable time, the structure of a space vehicle should include appropriate shielding.

Weightlessness. Before the advent of ballistic or orbital flights in manned space vehicles, the state of weightlessness was created artificially for short periods in an aircraft flying in a parabolic path. The results of space flight indicate that weightlessness has no appreciable effect on the physiological condition of astronauts. However, certain changes do arise as a result of a long period in the weightless state. Of the questions associated with human activity in conditions of long-term weightlessness, two are important: the effect of weightlessness on the cardio-vascular system, and the effect of weightlessness on the bone and muscular system.

The life-support system is designed to provide normal human activity and capacity in space flight. It consists of an air conditioning system and systems for providing humidity and removing excretions. In addition, it contains equipment for monitoring and controlling the system.

The process of creating an optimum life-support system includes a choice of the best systems for a specific mission and their combination into the total system.

The atmosphere conditioning system controls the gas composition, temperature, pressure and humidity of the cabin air and in the /41
astronauts' space suits. The control of the chemical composition of the cabin air is accomplished by eliminating carbon dioxide and harmful impurities liberated by the human being and by the cabin equipment, and also by supplying oxygen to compensate for its leakage and depletion via exchange with matter.

The moisture-supply system consists of equipment to provide the astronauts with water for drinking and washing. Since the water is a very heavy substance, it is desirable, in the interests of economizing the mass of the space vehicle, to use a system of water regeneration. Another consideration is that it is desirable to regenerate water from the humidity control system, and moisture and water intended for hygienic purposes. The water from the humidity control system has excellent chemical properties. However, as is pointed out in the literature abroad, tests have shown that the probability of a microbiological burden in this water is greater than in water obtained from fuel elements. Water obtained by regeneration of moisture is close in its chemical property to water obtained by regenerating water for hygienic purposes, and the facilities for the regeneration are structurally similar, since in both cases one needs to eliminate microorganisms and organic and inorganic impurities from the water.

The waste-elimination system. The simplest form of waste-elimination system is inboard tanks for gathering the waste.

A system for processing waste is designed to collect, store, and eliminate human waste and food waste. It is also necessary, as was a system for providing breathing air for the astronauts. The problem in storage of waste is the formation of gases resulting from exchange of matter in the microorganisms.

At present, the following methods are used for processing waste: refrigeration, heat sterilization, dehydration, combustion, and chemical processing.

In order to design a life-support system for astronauts, one needs to know standards for the exchange of matter, the rate of formation of carbon dioxide, the rate of consumption of oxygen, the amount of biological waste emitted, the requirements for

drinking and sanitary water, the atmospheric pressure, the temperature and humidity of the air, and the composition of the atmosphere, including the maximum allowable limit of contamination.

The oxygen supply. The required stock of oxygen aboard a space vehicle is determined by the number of the crew, the loss due to leakage from the cabin, and the requirements for refilling the cabin following venting. The losses due to leakage depend to a considerable extent on the structure of the vehicle. Usually multiple refilling of the cabin is planned on a mission. Depressurization is envisioned when the astronauts leave the space vehicle for an excursion in space, and also as a means of extinguishing a fire or removing toxic material from the cabin air. The gas (or liquid) for the air conditioning can be stored under /42 high pressure at the temperature of the surrounding medium, at low or intermediate pressure (cryogenic storage) and in the form of chemical compounds. The advantage of high-pressure storage is the high reliability. However, tests of the operation of manned aircraft have shown that cryogenic storage of oxygen and nitrogen has an advantage compared to storage under high pressure, since the high density of the liquid allows the volume to be reduced, for the same capacity, and therefore the mass of the bottle to be reduced. On the other hand, the cryogenic storage system as applied to space vehicles has two defects. First, because of the low temperature of the liquid, the bottles are sensitive to heat transfer to the surrounding medium. Secondly, in conditions of weightlessness it is difficult to draw the gas from the bottle.

As the duration of space flights increase, it becomes desirable to regenerate oxygen from carbon dioxide and water. However, one can use such a system only in space vehicles which use solar or nuclear sources of electrical energy, since none

of the electrical sources generated by chemical compounds has an acceptable specific capacity per unit mass.

<u>Temperature control</u>. The temperature in the space vehicle cabin is kept constant at the assigned level by supply or removal of heat.

The thermal load acting on the thermal control system consists of the heat liberated by the astronauts' bodies and by equipment, and the heat incident in the form of solar and planetary radiation. When a landing is made, the system is further loaded by atmospheric entry heating. In most cases, the control system must remove heat to maintain thermal equilibrium. In fact, one can imagine a space vehicle where there will always be an excess of heat to be removed from the cabin by the thermal control system. Then the construction of the thermal control system will be considerably simplified.

For space vehicle thermal control systems, one can use coolants, radiators and refrigeration cycles.

For most space flights the thermal control requirements are satisfied by a system with a cooling radiator, which has the advantage of high reliability, simplicity, and low mass.

As regards control of air humidity, it turns out that the most desirable method of control is cooling of a gas stream down to a temperature below the dew-point and collecting the condensate. /43
There are several methods of removing water from air. The simplest is used in the Mercury satellites. A sponge is located in the path of the air stream. Water accumulating in it is periodically squeezed out by means of a piston and flows down through a funnel into a special tank. On the Gemini and Apollo spacecraft, a wick-type separator was used to remove the water.

A certain concentration of carbon dioxide must also be maintained in the cabin of a space vehicle. On the flights of the Mercury, Gemini and Apollo spacecraft, absorption systems with lithium hydroxide were used for this purpose. Lithium hydroxide is a simple and reliable agent for absorption of carbon dioxide in the presence of water vapor in an air stream. For long-duration flights in space, it is desirable to use a regenerative system to eliminate carbon dioxide gas.

The matter of air purity for a space vehicle is very important, since the astronauts spend their entire time in the limited volume of a cabin. To control contamination of the air, one must reduce the number of potential sources of contamination. With this in mind, when designing the vehicle one should optimally locate the tanks, bottles, and working hardware, and also use only non-metallic materials which have successfully passed tests on the amount and kind of decomposition products under space flight conditions in a vehicle.

Contamination is eliminated by washing the air used in space, by filtration, carbon absorption, and catalytic combustion.

The American spacecraft Mercury, Gemini, and Apollo used activated charcoal to purify the air. This substance is an efficient absorber for many organic materials, including hydrocarbons with medium and high boiling point, alcohol, ketones, aldehydes and other substances, as well as gases, including ozone.

Catalytic combustion is used primarily to control the content of carbon monoxide and hydrogen by oxidizing them to carbon dioxide and water. Incandescent catalyzing agents are very often used in air purification systems.

CHAPTER 2

LOADS ON A SPACE VEHICLE: SPACE VEHICLE AERODYNAMICS

LOADS ACTING ON A SPACE VEHICLE

The design of a space vehicle must take into account a large /44 number of special features of the function of the vehicle and structural limitations which are mainly determined by the requirements as to the best location of the various onboard systems.

The space vehicle must house the crew and all the systems required for a successful flight and aoomplishment of the mission program.

The structure of a space vehicle is fundamental. The various functional features of the structure and the limitations imposed on it must be considered. The structure must be divided into modules and have the required shape to possess the necessary aerodynamic characteristics to give the vehicle stability and control. The latter requirement was not very important for the first stages of flight, but becomes exceedingly important in the atmospheric entry stage, and also when the modules are housed in the launch vehicle. However, the division into modules has a decisive influence on the choice of the launch vehicle system. In particular, when the main vehicle module has a large mass, it is more appropriate to increase the launch vehicle mass and to protect the modules from large loads during the powered section of flight than to increase the mass of the module itself in order to counteract these loads. The structure must also enable the crew members to enter and leave the modules, and to have access to equipment for purposes of repair or replacement.

An important requirement as regards the structure is that the vehicle should be strong enough when various factors act on it during the flight. These effects are most important during the launch. In addition, the structure must accommodate the mating of the spacecraft to the launch vehicle and should eliminate large disturbances when the spacecraft separates from the launch vehicle. Any unfavorable effect of the operation of the launch vehicle engines on the spacecraft should be eliminated in the design stage, including the possible effect of simply increasing the mass. The structure of the vehicle must also ensure meteorological and radiative shielding of the spacecraft elements.

The interaction between the structure and the various onboard systems is important in the choice of structural arrangement and system design. In particular: /45

a) the communication antennas must be shielded from heating while traversing the dense layers of the atmosphere. For efficient operation of antennas in deep space one must use materials which do not form compounds with carbon, and have low screening characteristics. Another possible solution for the matter of shielding is to unfold the antennas only when the space vehicle has been injected into its flight trajectory;

b) the device for deceleration in the atmosphere, for example the parachute or paraplane, must be protected during the stages of launch, flight in deep space, initial atmospheric entry, and subsequently, until the time of its use, and it must retain the necessary strength characterisitcs;

c) the control system and navigation equipment requires very accurate tuning, and therefore the chief requirement for it is high rigidity of all the structural elements, in spite of

their low mass;

d) the mechanism for docking in space must be flexible and appropriately damped. After docking is accomplished, the mechanism must be rigid and relatively strong, to avoid undesirable displacement of the structure during changes of altitude and while carrying out maneuvers.

The main accelerations act on the structure in the launch phase, especially during operation of the first stage engine, when the aerodynamic loads are maximal. As an example we consider the transverse cross section in the forward part of the equipment module; we assume that the structure in this section has the form of a circular arc of radius r. If an axial force P and a bending moment M act in this section, the axial load N (Figure 14) can be calculated from the formula

$$N = \frac{P}{2\pi r} \pm \frac{M}{\pi r^2}.$$

The load is made up of several components. The first component A arises from acceleration of the object. The spacecraft mass in this stage remains constant, while the mass of the launch vehicle decreases as the fuel is burnt. In essence the thrust here remains constant. The acceleration of the object increases. At the end of the first-stage burn the thrust disappears, and then the process is repeated at the start of the second-stage burn. The second component of the load D is due to the aerodynamic drag. This part of the load depends only on the forebody drag, which depends on the shape of the object, the flight speed, and the atmospheric density. The maximum value of the drag occurs at the time of the maximum dynamic pressure $\frac{\rho V^2}{2}$.

The third component of the load α is due to the angle of attack. An angle of attack arises during a programmed change in the pitch angle and when the space vehicle experiences wind gusts.

The maximum value of this load component ordinarily occurs also /46
at the time of maximum dynamic pressure.

The fourth component of the load G arises when wind gusts act on the space vehicle. There can be large discrete gusts or small periodic repetitive gusts. The most hazardous for the structure are small periodic repetitive wind gusts acting with a frequency equal to one of the natural frequencies of the body. The load on the spacecraft structure from these wind gusts must certainly not reach a maximum value during the dynamic pressure maximum.

An important source of loading is the large inertial forces which arise during incorrect operation of the control system for the launch vehicle engine, when all the moveable engines are deflected to an extreme position at maximum rate. Such a deflection of the motor causes large angular accelerations, accompanied by large inertial loads. Because of the large rate of application of the lateral thrust component, lateral bending oscillations can occur which apply an additional bending moment to the structure. The size of these loads increases as the launch vehicle mass decreases, and reaches a maximum at the end of the first-stage burn. It is not always required that the structure withstand this bending moment for an extensive period of time. However, the structure should retain its strength until the system for detecting malfunctions and the emergency rescue system for the crew module
has separated it. /47

During the ascent, the structure temperature reaches quite high values. The maximum dynamic pressure occurs approximately in the middle of the burn section. At this time the Mach number M of the free stream is a little above 1, and the aerodynamic heating is not large. Thus, when the aerodynamic loads are maximum, the temperature of the structure is small. During flight,

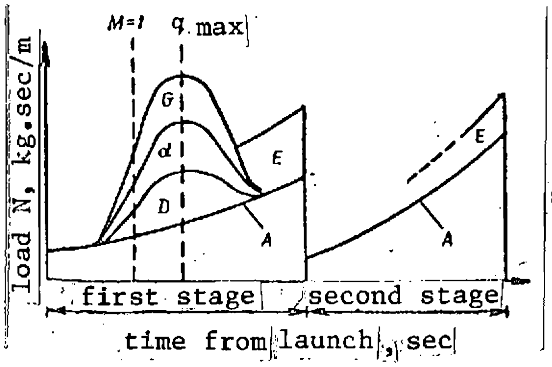

Figure 14. Variation of the loads acting on the structure with time (during the powered flight of the launch vehicle):

$$N = \frac{P}{2\pi r} + \frac{M}{\pi r^2}$$

(for a cylinder); A- acceleration; D- drag; α- angle of attack; G- wind gust; E- off-design deflection of the motor.

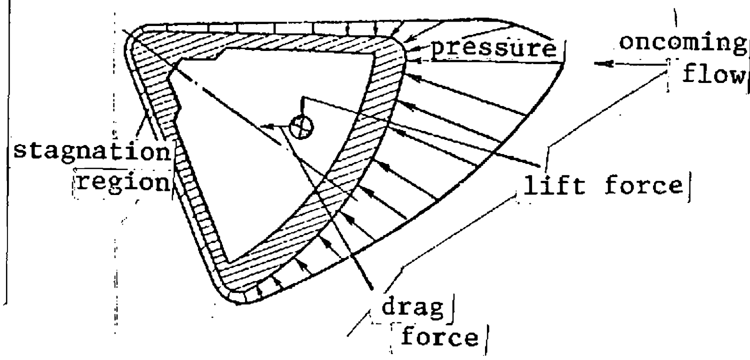

Figure 15. Pressure distribution over surface of spacecraft.

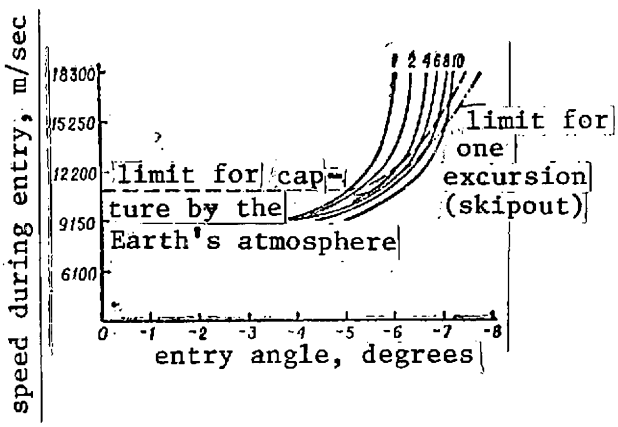

Figure 16. Loads on a spacecraft during ballistic entry into the dense layers of the atmosphere, as a function of flight speed and angle of the trajectory (1, 2, 4, 6, 8, 10 - values of allowable loading).

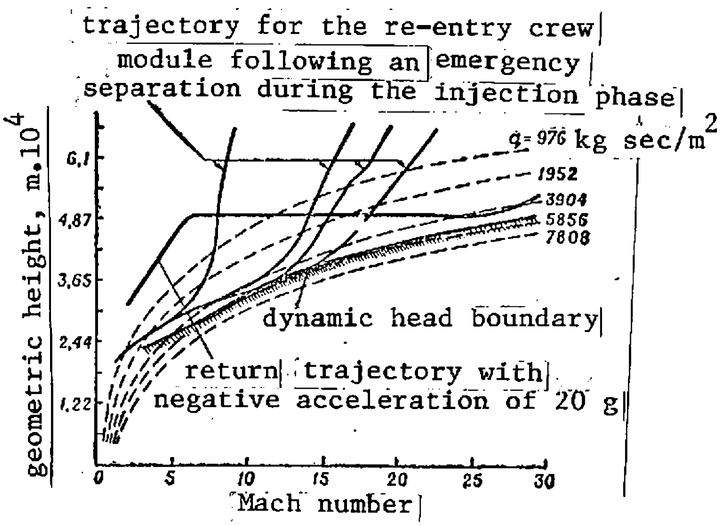

Figure 17. Atmospheric entry corridor.

the heat flux increases and reaches a maximum value later than the maximum of dynamic pressure. The time interval between these two maxima is 20-30 sec. The increase in temperature of the structure itself lags behind the increase in the heat flux to the surface. The temperature reaches a maximum at approximately the end of the first-stage burn, and slowly drops during the second-stage burn. An increase in temperature occurs when there are large loads due to accelerations and to a sudden deflection of the moveable motors.

Another important source of loading is turbulence due to flow over the body nose. Since the aerodynamic drag losses are small, the shape of the body nose need not be chosen to give minimum frontal drag, but can be blunted in order to decrease the heat flux: however, this makes the stream turbulent and generates large changes of pressure around the outside of the spacecraft housing.

It should be noted that there is one more form of loading acting on the structure, during emergency separation of the crew module. The magnitude of these loads is a maximum if there is emergency separation at low altitude, and these loads act only on the crew module. When the module separates, it experiences large accelerations during the emergency rescue system burn which ejects the module. At the end of this burn the module experiences forces due to very large frontal drag and large angles of attack. The asymmetry of pressure distribution acting on the module at this time is also taken into account in the design.

The loads on the space vehicle during atmospheric entry are due to the pressure of the air passing over the vehicle surface (Figure 15).

The resultant force passes through the vehicle center of gravity and therefore the vehicle does not experience any significant angular acceleration. In particular, it is stated in the foreign literature that the Apollo spacecraft has a lift-to-drag ratio of about 0.5 and can reduce this figure by rotation relative to its longitudinal axis. Therefore, the vehicle can vary its entry trajectory into the dense layers of the atmosphere, within certain limits.

In a return from the Moon, the crew compartment experiences a longitudinal negative acceleration of about 10 g in atmospheric re-entry, which can reach 20 g in emergencies. If we assume that /48 the atmospheric entry trajectory has a shape of a tube or corridor (Figure 16), we can estimate combinations of speed, flight height and corresponding dynamic pressures. For example, during negative accelerations of 20 g, the maximum load is encountered at a dynamic pressure $q \approx 4400$ kgsec/cm^2.

We can also calculate the conditions for atmospheric entry (Figure 17) during emergency separation between the end of the first-stage burn and injection into orbit. For this case it is assumed that the control system operates in an emergency mode, i.e., it provides continuous rotation of the module at a rate for which the lift force is zero. Then in certain cases there is a larger dynamic pressure than in the most unfavorable cases during lunar return. The flight of the module with constant lift-to-drag ratio occurs with approximately constant angle of attack and pressure distribution during the whole range of hypersonic speed. The acceleration is linear and the load depends on the pressure, /49 which depends in turn on the dynamic head.

The crew module. One of the main functions of the crew module (Figure 18) is to provide conditions required for life-support. As has been mentioned, the structure must afford pro-

tection from meteorites, and at least partially, from radiation. The main question in the design of the crew module is to ensure sealing of the module and to maintain the required environment within it. An important matter is to maintain the necessary temperature in the module. Since the heat loads during atmospheric entry are large, the heat shields are usually separate from the cabin structure.

The module is usually of welded construction, as dictated by the high sealing requirements. The most difficult design matter for this kind of structure is the action of external pressure during the period of the maximum dynamic head in ordinary flight, in the event of explosion of the launch vehicle, and when there is an asymmetric pressure distribution during emergency separation at low altitude. In designing a structure capable of withstanding these pressures, the designer begins with a monocoque form with a multitude of annular stiffening elements. However, for the above conditions, such a large number of annular elements is required that it is impossible to use a monocoque structure. Therefore, a sandwich type of structure is often used in practice for the module.

For the outer structure, the decisive factor is the external pressure. In addition, the outside structure has to carry the heat shield and must have a minimum reaction to the variations in pressure. This sometimes leads to a layered structure being chosen. /50

It should be noted that the outside structure can go up to 2800°C and more in the flight section following atmospheric entry, while the temperature of the module body will be less. /51

In constructing a space vehicle, one must also take into account the effect of low temperatures in deep space. The tem-

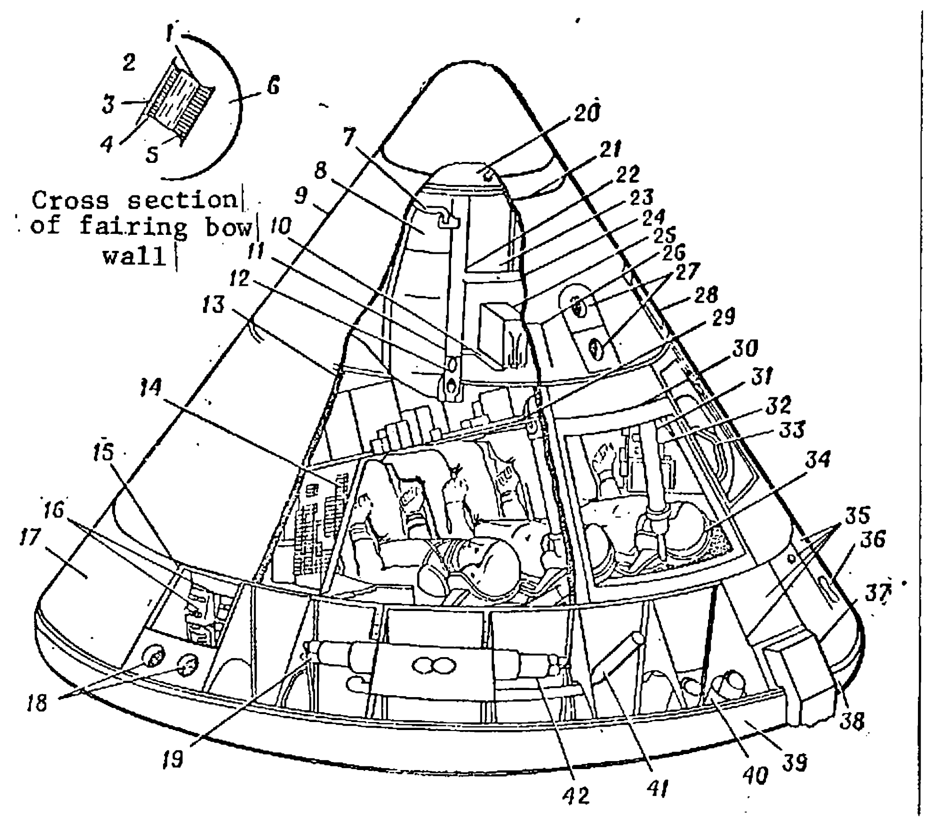

Figure 18. Composition of the crew module:

1- heat shield; 2- outer space; 3- ablative material; 4- stainless steel honeycomb; 5- aluminum honeycomb structure; 6- cabin; 7- bracket for parachute; 8- main parachute; 9- heat shield on forward part of module; 10- forward bulkhead of module; 11- retaining bolt for forward section heat shield; 12- retaining bolt for the emergency rescue system tower; 13- joint in the forward heat shield; 14- the left instrument panel; 15- service panel; 16- pressure control panel; 17- aft space for tools; 18; control motor for yaw angle, jet control system; 19- roll control motor of the jet control system; 20- forward entrance hatch; 21- fairing joint; 22- bracket; 23- the forward lock; 24- attachment for braking parachute; 25- program unit; 26- forward service panel; 27- pitch angle control motor, jet control system; 28- socket for emergency rescue system release support, during launch phase; 29- main instrument console; 30- crew entrance lock; 31- astronauts couch shock absorber; 32- right hand instrument console; 33- rendezvous observation lamp; 34- crew couch; 35- service panels; 36- roll angle jet control motor; 37- separating joint between crew and equipment modules; 38- joint in lower section of heat shield; 39- lower part of heat shield; 40- pitch angle motor of jet control system; 41- evaporator fan of air conditioning system; 42- roll angle of jet control system.

perature of the ablative material and the outside skin of the vehicle can be as low as -120C. The difference in thermal expansion coefficients of two materials can create considerable thermal stresses, and therefore, measures must be taken to eliminate thermal strain.

The equipment module of the vehicle contains all the equipmentment and power supplies which should not be located in the crew module. This module is mated to the launch vehicle, and therefore is the strongest structure of the vehicle. This module takes the form of a cylindrical shell with internal structures to carry the systems and the equipment. In the choice of the skin shape one should calculate the minimum mass that can withstand the alunch loads and pressure fulctuations and can also provide protection from meteorites.

The main form of possible disintegration of this kind of shell due to launch loads is loss of stability in the event of longitudinal bending.

A factor which has an important influence on the choice of the form of construction of the shell is the pressure variation. The reaction of the shell, and therefore, the stresses arising, depend to a considerable extent on the pressure and its distribution and timewise correlation with the modes of oscillations of the shell. In this event the geometry of the body nose is important.

A layered form of construction with porous low-density plastic filler and a layered construction with honeycomb filling have the best properties. Engineering designers abroad consider that a layered construction with discrete stiffening supports has the best weight characteristics for contemporary space vehicles.

The duration of a space vehicle flight in the dense atmospheric layers is small compared with the total flight time, but a good deal of time must be spent in designing and developing a manned space vehicle which will survive atmospheric flight. A spacecraft flying in the Earth's atmosphere experiences aerodynamic forces and moments, and these will be the main design loads acting on the spacecraft.

As is well known, the investigation of space by means of manned spacecraft has gradually encountered problems of increasing complexity. For example, in injection into orbit around the Earth, the satellite spacecraft were accelerated to orbital speed. /52
The flight took place in orbit around the Earth and subsequently there was deceleration for the return to Earth. The flight of a spacecraft with a landing on the Moon required acceleration to a speed close to the second cosmic speed, during the flight to the Moon and from the Moon to the Earth, and retardation for the return to Earth.

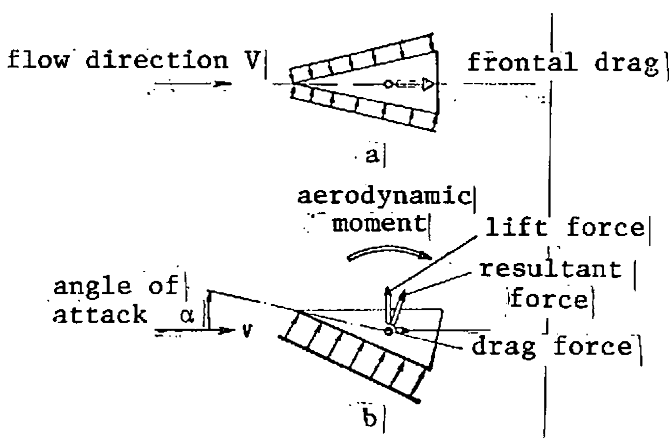

Figure 19: Pressure distribution on a wedge at hypersonic speed:
a) at zero angle of attack; b) at angle of attack α.

The next problem may be a flight to Mars. Here acceleration is required to greater speeds, and the Earth re-entry speed will be from 13,600 to 21,500 m/sec. For all flights the conditions in the injection section are roughly the same and vary only slightly, depending on the launch vehicle characteristics. However, the conditions at atmospheric re-entry depend on the entry speed.

The launch vehicle passes through a speed range of from zero to hypersonic in the atmosphere. In atmospheric re-entry the speed of the returning vehicle varies from the first and second cosmic speed down to subsonic speed.

In determining the flight characteristics of a spacecraft, the main factors are the frontal drag and the stability. Here one should remember that any body moving in a gaseous medium like air experiences pressure forces whose magnitude depends on the gas density, the spacecraft shape, and the flight speed.

Figure 19 shows flow over a wedge at zero angle of attack, and also at a certain angle of attack α, and also shows the pressure distribution acting on the wedge surface. A pressure distribution like this is typical for hypersonic flight speed with an attached shock wave. /53

By summing the pressure distribution over the body surface, one can determine the components of the forces acting in the direction of the velocity vector of the incident stream and perpendicular to it. The resultant of these forces is applied at a point defined to be the center of pressure.

By summing (actually, integrating) the moments of the pressure forces relative to a specific point (for example, the center of gravity) one can determine the total aerodynamic moment

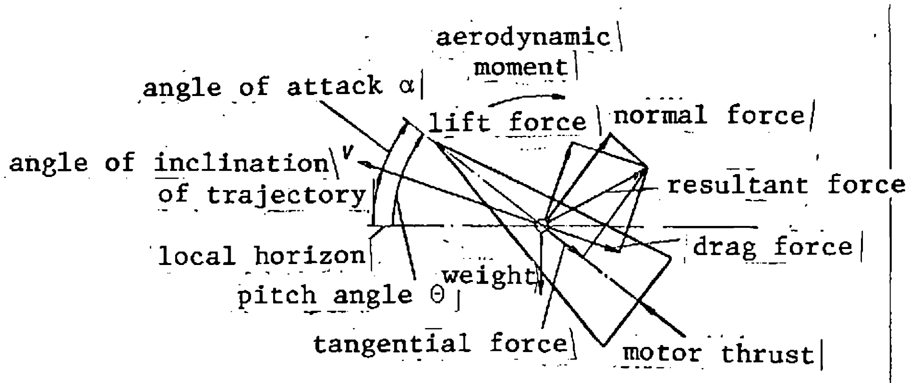

Figure 20. Coordinate system.

relative to this point.

Frontal drag. To examine the frontal drag and the spacecraft stability one usually introduces a system of coordinates. Figure 20 shows the body-fixed and the velocity coordinate systems. In the body-fixed coordinate system the resultant aerodynamic force reduces to a force directed along the body axis of symmetry, and a force perpendicular to it, while in the velocity system the force reduces to a force directed along the incident velocity vector, and a force perpendicular to it. In both systems the aerodynamic moment and the angle of attack are defined indentically.

The dominant forces composing the total frontal drag are the wave resistance (resistance of the vehicle nose section), the base drag, and the interference drag. A long thin body has a surface friction drag which cannot be neglected. The value of each of these forces varies according to the shape and size of the vehicle, the flight velocity and the angle of attack. In particular, the more blunted the vehicle shape, the larger is the wave drag; the larger the flight speed, the smaller the base drag, and all these components depend on the angle of attack. We shall consider the aerodynamic forces acting on individual

items of the system.

Launch vehicle. The aerodynamic forces acting on the launch vehicle, just as on an aircraft, affect its flight characteristics. The engine thrust of a spacecraft must overcome the action of these forces during the entire flight. Each spacecraft has its own characteristic, and it is therefore impossible to make a direct comparison of the flight characteristics of an aircraft and a launch vehicle. For an aircraft, the dominant flight characteristic is often the flight range for a given payload, while for a launch vehicle it is the flight speed up to which it can accelerate a given payload. Therefore, for an aircraft at the steady flight speed, the thrust must be equal to the frontal drag, while the aerodynamic lift is equal to the weight. For a launch vehicle the thrust must be greater than the frontal drag and the weight, in order to accelerate the payload. /54

The frontal drag, the thrust and the weight affect the flight characteristics of a launch vehicle.

Elementary mathematical computations show that the velocity loss due to frontal drag is less by a factor of 15 than that due to gravity. However, as the spacecraft size increases, the drag becomes still less. The main reason for this is that the drag is a function of the spacecraft area, while the weight is a function of its volume. It is clear that as the engine thrust increases and the time of the burn decreases, one can reduce the speed loss due to gravity, since with an increase in acceleration we require less time in order to attain a given speed. Of course, this increases the frontal drag and the speed loss due to it. The best arrangement is the case where the sum of the speed loss due to drag and gravity will be a minimum.

To optimize the launch vehicle thrust we require a different approach. As the maximum dynamic pressure increases (due to an increase in the engine thrust) we must make the launch vehicle structure heavier, to withstand the increased aerodynamic loads. The result is to reduce the payload of the launch vehicle and increase the speed loss.

The shape of the module carrying the payload has a small effect on the launch vehicle flight characteristic, but this effect is considerable in comparison with other factors. The shape of the payload module can shift the point of application of the normal force forward, which leads to an increase in the bending moments, and thus, to an increase in the structural mass. The presence of sharp edges (or corners) in the module is also undesirable: this can lead to the appearance of vibrational loads which can excite bending oscillations of the launch vehicle, and even destroy the integrity of the structure. The presence of protruberances on the rocket may cause breakdown of a structure because local vibrational shock waves and secondary flows are generated. All these phenomena cause an unnecessary increase in the structural mass of the launch vehicle in order to increase the strength. /55

The separable emergency stage (emergency rescue system, ERS). The separation of the emergency stage from the space system can be accomplished by means of a high-thrust rocket engine, located in front of the spacecraft, and pulling it away from the launch vehicle. To be specific, in the majority of space vehicles the ERS rocket motor is in front of the spacecraft.

We now consider the requirements for the rocket motor of this system. In analyzing the injection trajectory one can see that the maximum frontal drag occurs roughly at the point in the trajectory where the maximum dynamic pressure is generated. At

this point the thrust of the emergency engine must overcome the drag so that the spacecraft can separate from the launch-vehicle. The drag coefficient of the emergency stage is roughly that of the launch-vehicle, while the area of cross section of the separable stage is usually somewhat less.

An important parameter is the mass, and the mass of the emergency separable stage is less by a factor of 100 than that of the launch vehicle. This means that the effect of drag, the drag per unit mass of the spacecraft, is much less for the emergency stage.

The amount of fuel required is determined by the energy expended in lifting the emergency stage to a safe height when the emergency occurs during the launch. A safe height must be reached also to make it possible to use landing devices during the descent. The engine must ejct the spacecraft to a safe distance in the lateral direction to avoid its descending on the burning launch-vehicle. The emergency rescue engine (Figure 21) must operate at heights where its thrust should be restricted to a level which does not cause an increase in the allowable loading for the crew, since there is no frontal drag to reduce the vehicle acceleration at large heights.

The emergency rescue system engine is mounted in front of the spacecraft and its rocket engine greatly changes the flow field washing the spacecraft. Therefore the drag characteristics of the spacecraft are greatly changed during the engine burn. The jet diameter depends on the static pressure or on the height at which the emergency separation takes place. The frontal drag of the spacecraft returning to Earth increases because of the effect of this jet. At low altitudes the jet is of small diameter and, because of its high speed, acts as an ejector, which accelerates the air at a higher pressure than that of the

surrounding medium, increasing the drag of the descending spacecraft. At high altitude where the jet expands rapidly, the spacecraft shades the jet from the effect of the oncoming stream, and the drag is less. /56

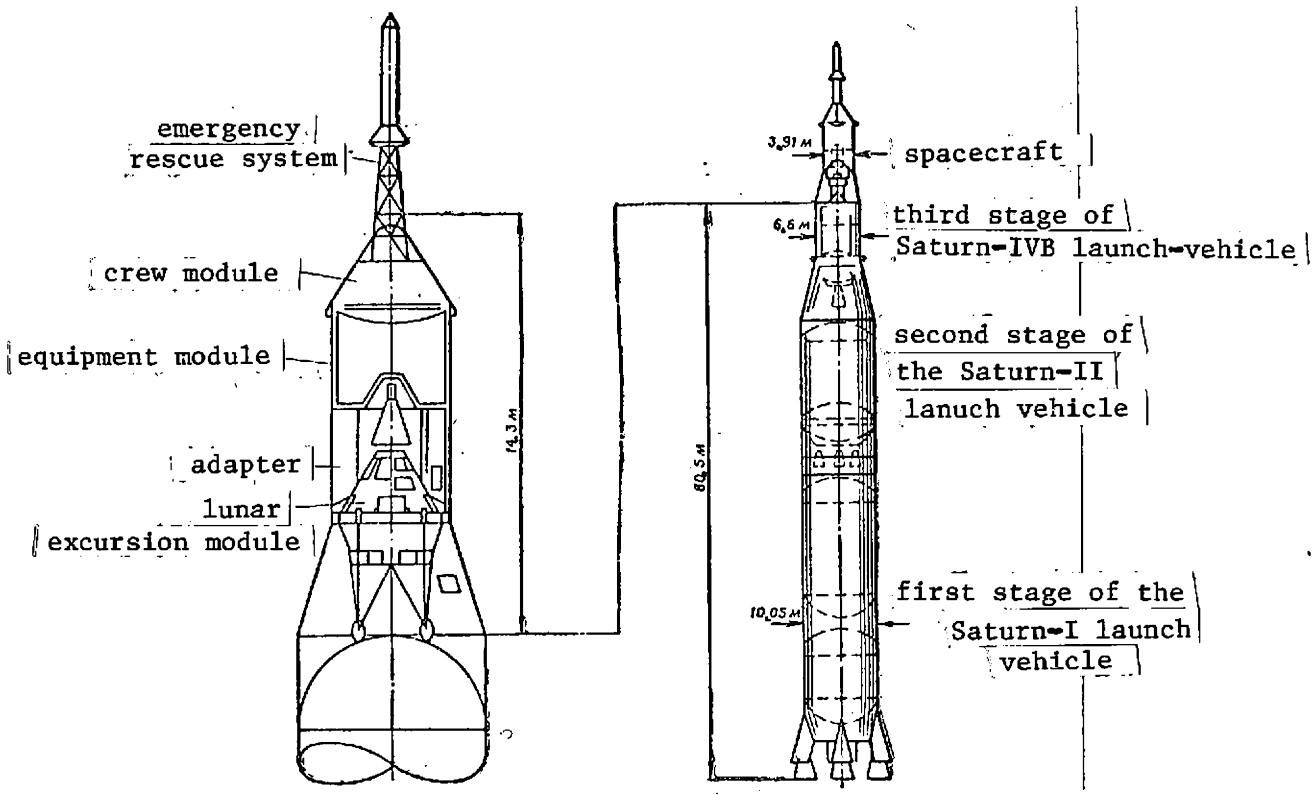

Figure 21. Launch vehicle for the Apollo spacecraft and the emergency rescue system.

Atmospheric re-entry of a space vehicle. The effect of the drag on the re-entering vehicle is most important in the section where the vehicle enters the dense atmospheric layers. The drag then arising reduces the vehicle speed to a small subsonic speed. Deceleration due to aerodynamic forces has a considerable advantage over deceleration due to a motor.

As space investigation progresses, the atmospheric re-entry speeds are increasing. These speeds vary from an orbital speed

of 7.91 km/sec and a lunar return speed of 11.2 km/sec up to planetary return speeds of 13.9-21.35 km/sec. The main factor affecting deceleration of a vehicle in atmospheric entry is the ballistic coefficient.

The ballistic coefficient of a system, $W/c_0 S$ (the weight W divided by the drag coefficient c_0 multiplied by the original area S) is the main factor determining the height at which the maximum deceleration of the re-entry vehicle occurs. It is assumed that the flight trajectory is a straight line, and that the atmospheric density varies exponentially. /57

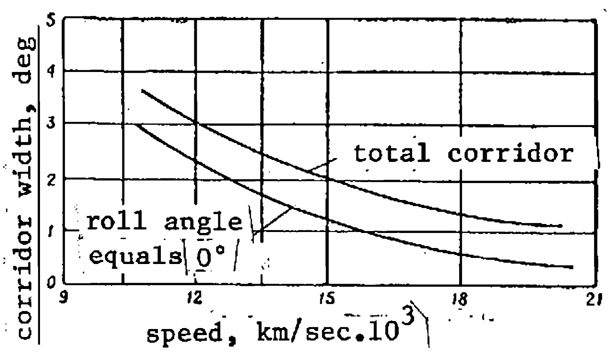

Figure 22. Effect of entry speed on the entry corridor width (lift -to-drag ratio of 1; dynamic pressure q = 488 kg.sec/m^2).

A corridor for ballistic atmospheric entry can be drawn as in Figure 22, with the entry angle (corridor width) as the ordinate, and the entry speed as the abscissa. The deceleration limit during re-entry of manned vehicles is usually taken as 10 g, and can approach 20 g in emergency conditions. Loads like these can be met with not only in a normal entry, but also when the vehicle approaches the atmosphere along a skipout trajectory, where the speeds are very great and the speed does not drop below the second cosmic speed. Therefore, a second limit is required. Figure 16 shows two broken lines, one of which determines atmospheric entry when "capture" of the returning vehicle occurs (speed drops below the second cosmic speed), and the second line shows atmospheric entry with one skipout. Atmospheric entry in the section to the right and below the broken lines satisfies the boundary conditions. The area between an entry angle satisfying these limits and an entry angle corresponding to the chosen maximum deceleration is the

entry corridor. It can be seen that, for a maximum allowable deceleration of 10 g, and the boundary condition that the speed be less than the second cosmic speed, the entry corridor falls to zero at a speed of 15.25 km/sec. This means that for an entry at speed above 15.25 km/sec and a deceleration equal to 10 g, the vehicle speed cannot drop below the second cosmic speed. Several methods exist for reducing the drag load and increasing the corridor width. First, the frontal drag can be increased by increasing the area during atmospheric entry, and therefore it is possible to control the aerodynamic loads to achieve a wider entry angle corridor. Another, more effective method is to use lift. /58

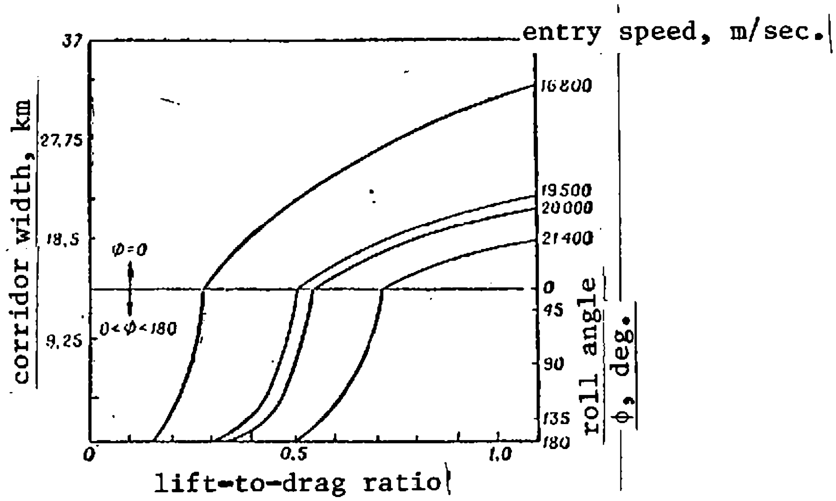

Figure 23. Effect of lift on atmospheric entry corridor.

We recall that when discussing frontal drag it was stated that the total (integrated) pressure distribution over the vehicle surface gives only the frontal drag force directed along the velocity vector. If there is asymmetry of the flow over a symmetric body at a certain angle of attack or in flow over an asymmetrical body, the total pressure distribution generates a lift force in the direction perpendicular to the velocity vector. The effect of lift on the entry corridor is conventionally de-

scribed in terms of the lift-to-drag ratio. The presence of lift makes the vehicle follow a curved trajectory. In this case, for given deceleration limits, a vehicle using lift will enter the dense region of the atmosphere at a steeper angle than a vehicle with zero lift. In the same way, aerodynamic forces can be used to achieve atmospheric capture of a re-entry vehicle at a more shallow entry. The effect of the lift force in increasing the entry corridor is shown in Figure 23. The boundaries of the entry corridor are the curve at the maximum allowed deceleration of 12 g, and a curve showing atmospheric entry with one skipout and return maneuver. A small lift force produces a considerable expansion of the entry corridor. /59

Landing devices have been designed, based on the use of drag. At present a parachute is used which reduces the terminal velocity of a returning vehicle to a value permitting a safe landing on Earth; it is purely a decelerating device. The operating principle is based on the great increase in drag area, which increases the ballistic coefficient W/c_0^S of the vehicle.

The static stability is the property of a body to create restoring forces or moments such as to return it to the equilibrium position when it departs from an equilibrium position in flight. Considering rotation of a vehicle relative to its center of gravity, we can regard the stability in a very simple sense as being determined by the sign and magnitude of the restoring aerodynamic moment when the vehicle departs from the balanced or neutral position. A moment which tends to return the vehicle to the original position is a stability moment, while the moment which tends to increase the departure from the original position is an unstable moment. The aerodynamic moment due to an aerodynamic force is applied at the center of pressure relative to a certain point (usually the vehicle center of gravity). There are three possible ways of changing the moment

characteristics: varying the magnitude of the force, displacing the center of pressure, and displacing the center of gravity. Usually the moment characteristics of a flight vehicle (or aircraft) are linear in the operating range of the angle of attack. The elements of an aircraft with a small expendable mass (e.g., with a certain amount of cargo and fuel) are so arranged that the center of gravity of the vehicle is displaced to a minimum degree as the mass is reduced. This displacement leads to changes in the aerodynamic moments which are compensated for by small equalizing devices which arrange for the aircraft to fly at an angle of attack corresponding to the best flight characteristics. The aircraft always possesses static stability in flight.

The launch vehicle. The requirements as regards static stability for a launch vehicle differ considerably from the corresponding requirements for an aircraft. The launch vehicle is launched with zero speed, and as it moves in the Earth's atmosphere, gradually increases its speed through subsonic, supersonic and hypersonic values. The problem in ensuring stability of the launch vehicle arises because of the large expenditure of mass due to burning of the fuel, which changes the position of the center of gravity. The fuel mass of a launch vehicle constitutes 85-90% of the launch mass. The rate at which mass is lost determines the requirements for control, which must operate during the whole time of the burn, i.e., from the moment of launch, when the aerodynamic loads are zero, up to the beginning of flight at high altitudes, when the aerodynamic forces again return to zero. Therefore, control of a launch vehicle by aerodynamic forces alone is insufficient. Control of a launch vehicle is accomplished by controlling the thrust vector. A moment /60 (created by the thrust) is used to vary the attitude of the vehicle in a given direction.

The separable emergency stage must be designed for the same flight conditions as the launch vehicle. However, the require-

ments for its stability differ somewhat from those of the vehicle. It is desirable to design the vehicle to ensure stability by passive stabilization, i.e., it must have static stability throughout the whole range of flight velocity. In an emergency separation during the section when large dynamic pressures are present, the separating emergency stage must fly in the ambient range of altitude in such a way that the crew does not experience undesirable accelerations due to aerodynamic loads. It is stated in the foreign literature that if the Apollo spacecraft began to spin following emergency separation at the maximum dynamic pressure, transverse loads of 10-14 g could arise. These values considerably exceed the allowable physiological limit for human survival. However, side forces must be generated to eject the spacecraft from the launch vehicle. This is done by aerodynamic methods or a side component is generated in the thrust vector of the emergency rescue system motor. Since the initial flight velocity of the launch vehicle is small, aerodynamic methods of creating the side component are ineffective. The fuel mass for the emergency rescue system motor is about 20% of the total mass of the separating stage, and therefore, a considerable shift in the center of gravity occurs during the burn. This naturally complicates the matter of static stability. The variation in the required degree of stability can be achieved aerodynamically (by displacing the center of pressure rearwards) or by moving the center of gravity forward, by adding a centering load at the nose section of the motor of the emergency rescue system. The second method is normally used since it allows one to avoid aerodynamic surfaces within the body of the returning spacecraft, which in turn avoids the need to attach and then to separate these surfaces. It is normally more convenient to sue a balancing load from the point of view of the weight characteristics, since it is jettisoned along with the base when it is no longer needed, and is not injected into orbit. It is a very difficult problem to achieve stability of the separated rescue stage, and it has

been solved in the design of the manned space vehicle. The difficulty is due to the diversity of operating conditions, the complexity of the aerodynamic characteristics, and to the shift with time of the position of the center of gravity.

A re-entry vehicle must possess static stability during the atmospheric entry. Unlike the launch vehicle, it has no motor whose thrust could be used for control. It is unattractive from the viewpoint of the weight characteristics to include a motor in the re-entry vehicle for this purpose alone. However it is quite simple, in the opinion of the specialists, to develop a re-entry vehicle which would have the desired degree of static stability. Some vehicles of this type have the form of a body with blunted nose. These vehicles have a forward heat shield (having the shape of a spherical segment). Only a body of this kind has a constant position of the center of pressure when the angle of attack varies. All the pressure forces pass through the center of the sphere. A body with a blunted nose can have lift, but a spacecraft whose center of gravity lies on the body axis of symmetry is balanced at zero angle of attack and the lift is zero. By displacing the center of gravity in a vertical direction (along the axis of symmetry) by an appropriate mass distribution, one can balance the spacecraft to obtain a certain aerodynamic characteristic. Lift was created in this way in spacecraft of the Soyuz and Apollo types. The spacecraft enters the atmosphere at the balanced angle of attack with the balancing lift force and retains its attitude in space during the entire flight.

The lift created when the vehicle is balanced at a certain fixed altitude can be controlled by rotating the spacecraft relative to its longitudinal axis (i.e., rolling). At zero roll angle the spacecraft lift will be directed away from the Earth. When it is rolled, a lift force is created which is directed

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lateral to the flight trajectory. At a roll angle of 180 degrees the lift force is directed towards the Earth. The roll motion is achieved by small rocket motors located on the spacecraft body. If lift is not required, the spacecraft must rotate continuously. The result is that the trajectory is a spiral, and the lift force is equal to zero.

The static stability for this case is defined as the capacity of the body to create restoring moments when there is an angular displacement relative to the center of gravity.

The dynamic stability of a spacecraft is its capacity to create stabilizing moments, arising when there is an angular velocity during an oscillation of the spacecraft about its center of gravity. Since the air stream passes from the forward part of the spacecraft towards the rear in a certain time, because of the rotation and displacement of the spacecraft, the stream impinges on the rearward part of the spacecraft at a different angle of attack than on the forward part. Depending on the spacecraft shape and size and the angular velocity, this moment can be a stabilizing one or a destabilizing one.

Since the angular deviations of a launch-vehicle are small, the angular velocities associated with these are small. Therefore, the effect of dynamic stability on the motion of the launch-vehicle as a solid body is not significant. /62

The lift of the launch vehicle is always damped, and the changes in the dynamic head can act as a damping influence, and can also excite oscillations, but only to an insignificant degree. The launch vehicle control system can ensure good damping of the oscillation. The construction of the launch vehicle is very considerably affected by the fact that it is not a solid, but an elastic body. If an elastic launch vehicle, oscillating

relative to a node point, is dynamically unstable, then the excitation of natural frequencies of oscillation of the spacecraft can be so great that they cause the structure to breakup. When the frequency of undamped spacecraft oscillations is equal to the natural frequency of bending oscillations of the launch vehicle, a resonance occurs, and the launch vehicle begins to oscillate with the natural frequency of vibrational bending oscillations. The frequency of the aerodynamic oscillations depends on the dynamic head, which first changes from zero to its maximum value, and then drops back again to zero. Many kinds of oscillations exist at various frequencies, and these must all be taken into account in the design of the launch vehicle.

When considering the dynamic characteristics of the separable rescue stage, one must take into account that it is uncontrolled.

The dynamic stability characteristics of a spacecraft with an engine mounted on a truss structure are very sensitive to changes in aerodynamic shape. A hysteresis frequently occurs, since the rescue stage is essentially a body of cone-cylinder-flare type, for which this phenomenon is observed. The effect of lift reduces to damping of the oscillations, but since the rescue stage flies under conditions of decreasing dynamic head, apart from the short period when the rescue system motor is burning, a destabilizing moment arises. Therefore, one must carry out a detailed analysis of the dynamic motion of the free body to evaluate the effect of factors entering into the dynamic stability of the rescue stage.

The dynamic stability derivative of a re-entry spacecraft is usually stabilizing. However, for certain angles of attack the vehicle may be dynamically unstable. The re-entry vehicle is controlled by rocket motors, which stabilize the motion relative to the balanced attitude and must avoid any dynamic in-

stability occurring during the flight. Therefore, in flight under normal conditions the dynamic instability does not appreciably influence the controlled re-entry vehicle.

There are several methods for determining the aerodynamic characteristics of a manned spacecraft: one can calculate the aerodynamic coefficients by appropriate theories, one can test models in wind tunnels, and one can test free-flight models in actual flight conditions. These methods are used in the design and development of manned space vehicles. After the flight mission is defined, trajectories are calculated to determine the requirements of lift, drag, and stability. Limits as regards the make-up of the spacecraft arise with regard to the shape, size, and mass. Most of these constraints are due to the payload and the energy capability of the launch vehicle. /63

After the spacecraft shape is chosen by design calculations, a model is prepared which will undergo extensive tests in ground aerodynamic facilities. To obtain a detailed analysis of the composition and flight characteristics, tests are carried out over the whole range of flight speed and angle of attack. For the tests on the flight performance, models are made. During these tests the operation of the onboard systems is demonstrated and tests are made of the engineering solutions which underlie the design. If it becomes necessary to alter the structure during the flight-tests of models, an analytical investigation is done, and then new models are prepared and tested.

Manned re-entry vehicles can be subdivided into various types, depending on the aerodynamic characteristics, i.e., the lift-to-drag ratio. The effect of this ratio on the entry corridor, which is a measure of the maneuver capability of the vehicle, was considered in detail above.

CHAPTER 3

AERODYNAMIC HEATING AND THERMAL PROTECTION OF SPACECRAFT

As has already been mentioned, heat inputs from various heat sources act on a spacecraft in all sections of the flight trajectory, and during the launch and atmospheric entry the spacecraft is subject to very strong aerodynamic heating. /64

AERODYNAMIC HEATING

In order to return safely to the Earth, a spacecraft must dissipate the kinetic and potential energy accumulated relative to the Earth's surface. For existing manned spacecraft, a good means of dissipating a large part of this energy without adding to the spacecraft mass is aerodynamic braking in the dense layers of the atmosphere, in which the kinetic energy of the spacecraft is transformed into thermal energy of the surrounding air. The central problem in aerodynamic braking is that the spacecraft absorbs a certain amount of the energy dissipated. Here the aerodynamic heating can be very intense, as evidenced by the fact that a high percentage of meteoritic bodies is burned up and disintegrated during atmospheric entry.

In order to analyze the qualitative laws of aerodynamic heating during atmospheric entry, one must consider the structure of the atmosphere and its parameters which affect the flight of a spacecraft.

The speed of a vehicle determines what volume of air is incident on the vehicle per unit cross-sectional area of the air stream per unit time. The mass of air flowing over the vehicle and incident per unit cross-sectional area of the vehicle per unit time is equal to the volume density multiplied by the

volume of incident air ρ V. This air mass flux, multiplied by the relative momentum of the air per unit mass, determines the momentum incident per unit area of the flow per unit time.

The kinetic energy incident in unit air mass is equal to $1/2\ V^2$. Therefore, the kinetic energy transmitted to unit sur- /65 face area of the vehicle is equal to $1/2\ \rho V^3$. This kinetic energy is an upper bound on the specific heat flux q reaching the surface of a re-entering vehicle. The integral of the quantity $1/2\ \rho V^3$ over the descent time through the atmosphere represents the work performed by the spacecraft in the complete deceleration, and is an upper bound on the total quantity of heat.

The characteristics of aerodynamic heating can vary considerably, depending on the flow passing over the vehicle. Flow regimes can be divided into two classes: first, flow of rarefied air of low density at high altitude, which can be considered as a flow regime in which the vehicle surface interacts directly with each air particle, and secondly, flow of air of large density at low altitudes, which can be represented as a flow regime where the vehicle surface interacts with the air as if it were a continuous medium.

During atmospheric entry, a vehicle first encounters only individual air particles. In free-molecular flow the air density is so small that collisions between air particles can be neglected, while collisions of particles with the vehicle surface cannot be neglected. At each such collision the vehicle loses part of its stored energy and some amount of momentum relative to the Earth, but if the collisions are elastic, the vehicle does not absorb thermal energy. For elastic reflection, the effect of energy dissipation on the vehicle surface is analogous to the effect upon reflection of a plane wave.

The density of particles at the vehicle surface can be greater by a factor of two than the density of the undistrubed air stream, since the mass flux of air reaching the vehicle surface is added to the reflected mass flux. The picture of the distribution of density of reflected particles is identical with the distribution in intensity of a reflected plane wave. If an air particle is scattered diffusely, then the inelastic collisions communicate to the surface a corresponding pulse of thermal energy. Then the energy of the air particles can be completely absorbed, or can first be absorbed, and then reradiated with less kinetic energy.

The process of collision of air particles with the vehicle surface is very complex. The nature of a collision is determined primarily by the surface properties, and also by the energy and momentum of the incident air stream. Since all high energy particles of the incident air stream can come into direct contact with the relatively cool vehicle surface, the kinetic energy $1/2\ \rho V^3$ absorbed by the vehicle surface will be a maximim under free-molecular flow conditions. Each incident particle can transmit its energy only to the vehicle surface, and not to other air particles. /66

However, the flux of energy reaching the vehicle surface is small in this regime, because of the extremely low air density.

A re-entering vehicle dwells in these conditions for a short time, and therefore its surface receives only a small amount of heat.

As the density of the incident stream increases, the air particles begin to collide with particles reflected from the surface or radiated by it. This mutual collision of particles

in a near free-molecular stream reduces the mean energy of collision of particles with the vehicle surface. Nevertheless, almost all the energy of the incident stream reaches the surface. However, a "buffer" zone where particles collide with one another reduces this part of the stream energy which is absorbed by the vehicle surface.

As the vehicle penetrates deeper into the atmosphere, the density of the incident air stream increases to a level where the reflected or re-radiated particles are compressed between the vehicle surface and the oncoming stream. This increases the density at the vehicle surface. The vehicle captures and releases the air particles (the "snow plow" principle). But the air stream is very energetic, since the vehicle communicates a relatively large arbitrarily distributed kinetic or thermal energy to the air particles. When the vehicle reaches the transition flow regime, a shock wave (a jump in density) is formed between the incident stream and the vehicle surface. The distance from the density shock wave to the vehicle surface depends, in particular, on the molecular mean free path of the air and the nose radius of the vehicle.

The bow shock formed ahead of the vehicle determines the sphere of influence of the vehicle on the air stream.

The effect of a shock wave on the density profile ahead of the vehicle is illustrated schematically in Figure 24. The shock wave makes it possible to transfer energy of the oncoming stream into an arbitrarily distributed thermal energy of the compressed air layer. The formation of a shock wave here is evidence that the medium has become a continuum. The term continuum means that the air, at least in certain respects, can be regarded as a continuous medium.

When the air stream density is high, the effect of its viscosity begins to appear at the vehicle surface in the region of a thin boundary layer. In other words, the friction forces at the vehicle surface affect only a layer of air located very close to the vehicle surface. In this high density layer, or boundary layer, in continuum flow, the air particles in contact with the surface cannot migrate relative to the surface.

The heating during atmospheric re-entry of a spacecraft in the continuum regime has a contribution from convective heating transmitted into the boundary layer, and from radiation from the high-temperature layer of the air located between the shock wave and the nose section of the vehicle. /67

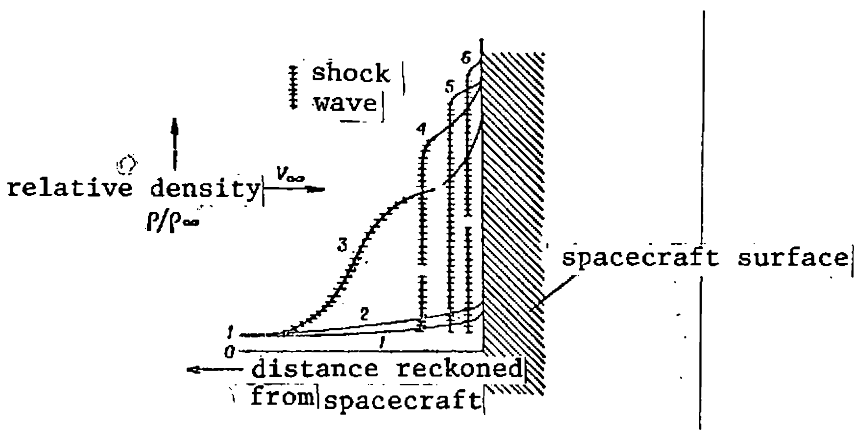

Figure 24.

Schematic representation of the density profile in the flow ahead of a space vehicle in different flow regimes:

1- free-molecular flow; 2- flow close to free-molecular; 3- fully developed flow with slip; 4- incipient flow with slip; 5- viscous layer; 6- boundary layer.

The flux of thermal radiation to the vehicle surface can be reduced by making the vehicle nose sharp. When there is flow over such a body the shock wave is located in the immediate vicinity of the nose and the amount of air with high enthalpy which transmits radiant energy to the vehicle surface is less than when

the nose section is blunt. However, the use of a sharp nose is directly at variance with the requirement to reduce the convection heat flux, and therefore a compromise solution is usually adopted.

The largest and smallest heat fluxes can differ by four orders of magnitude. Therefore, the requirements are: to control the vehicle temperature within limits such that the load on the cooling systems should be a minimum; to keep the heating of the heat shield system used during atmospheric re-entry below a level at which the material might change its state, thereby reducing its efficiency; and to provide heat protection for the fuel tank and equipment which do not have thermal control systems.

In the injection section, when the specific heat flux reaches a large value, the main problem is to avoid damage to the thermal protection and control systems, which often take the form of quite thin surface layers and are used in the atmospheric re- /68 entry. During re-entry the heat flux increases by a factor of 100 in comparison with its value during injection. The heat flux is so large that the main problem is to keep the temperatures of the main structural elements to the allowable operating level. In addition, it is also necessary to investigate methods to reduce the amount of heat penetrating to the inside compartments of the vehicle.

During its flight in deep space, a space vehicle is exposed to the action of a comparatively small heat flux arising from heat radiation from the Sun and planets of the Solar System. The surface of the thermal insulation covering the vehicle will radiate energy whose magnitude is proportional to the fourth power of the temperature of the blanket. Thermal equilibrium is reached when the solar radiant energy reaching the blanket equals the radiant energy radiated by the blanket surface. In

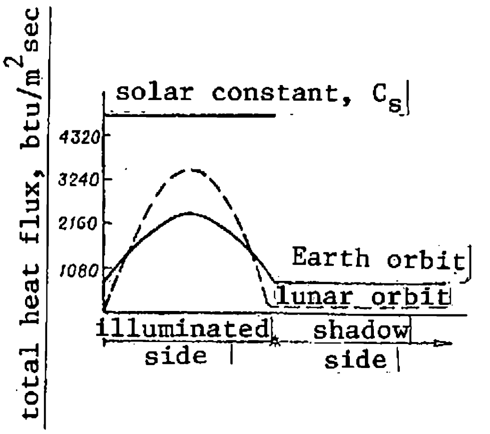

Figure 25.
Specific heat flux at the surface of a spacecraft in a low orbit flight.

analyzing the heat balance of a surface element perpendicular to the incident radiant flux, one uses the solar constant C_s (the thermal energy per unit area at the given point in space).

This constant depends on the distance from the Sun and varies for the orbits of the various planets. If a spacecraft flies close to a planet, one must also account for the energy and solar radiation reflected by the planet, and the energy of the self-emission of the planet. The values of these heat fluxes are shown in Figure 25 for lunar and Earth orbits (low orbits of altitude 80-100 km above the Earth's surface). On the shadow side of the Moon there is only the heat flux radiated from the surface of the Moon itself, which has a rather low mean temperature (about -120 to -130° C). The illuminated side of the Moon reflects roughly 7% of the solar energy incident on it; the heat is radiated from the lunar surface, at a temperature of 120° C. The Earth has considerably higher mean temperature on the shadow side than the Moon, but a considerably lower temperature on the side illuminated by the Sun. The energy reflected from the Earth is roughly 35% (its albedo) of that incident from the Sun. Therefore, the surface of a vehicle facing the Earth will receive less heat flux than the surface

turned towards the Sun. The straight line on Figure 25 shows the heat flux of direct solar radiation. /69

The emissivity of a vehicle surface determines not only the fraction of the planetary radiant energy absorbed, but also the amount of solar energy absorbed and reflected. The spacecraft designer chooses blanket materials with characteristics such that the supply and removal of heat can be balanced in order to maintain the desired temperature within the spacecraft. If sufficient power is stored, active cooling or heating of the surface can be provided. The use of a passive thermal control system allows the power requirements to be considerably reduced. The designer has various means at his disposal for the control of temperature: perfect absorbers, i.e., surfaces covered with black paint and absorbing energy at any wave length; perfect reflectors, i.e., polished surfaces covered with silver and reflecting energy of any wave length; solar absorbers, i.e., aluminized surfaces, absorbing energy only in the solar spectrum; and solar reflectors, i.e., surfaces covered with white paint and reflecting energy only in the solar spectrum. A possible additional method of temperature control is the use of thermal insulation materials. Ordinary thermal insulation materials such as fiberglass have quite good insulation characteristics at atmospheric pressure, and if the blanket takes the form of two structural layers with the space between them filled with this kind of insulation material, then, by controlling the pressure in this relatively small volume, the heat flux entering and leaving the spacecraft can be altered considerably.

In the flight section outside the atmosphere, the heat balance of a spacecraft can be achieved in most cases by choosing a coating which will allow passive thermal control. If this is not possible, then active and semi-active methods of thermal protection can be used.

Protection from kinetic heating in the atmosphere. The greater part of the kinetic energy during atmospheric deceleration of a spacecraft is dissipated in the wake behind the vehicle, and only a small part is transmitted to the vehicle in the form of heat energy. However, this small part of the energy can create thermal stress in the vehicle surface. As has been mentioned, the thermal conditions are determined to a considerable degree by the manner in which the vehicle penetrates the dense layers of the atmosphere. Large heat flux arises for steep entry trajectories. For shallow trajectories the heat flux is less, but since the flight duration increases, the total amount of heat received will be larger. Figure 26 shows the heat flux and the total amount of heat received in the flight of various spacecraft. For comparison, the heat flux from a nuclear explosion is shown. /70

HEAT SHIELDS

Radiating systems. In developing heat shields it might seem that the simplest thing is to use high-temperature material which could withstand the effect of very large heat fluxes in flight.

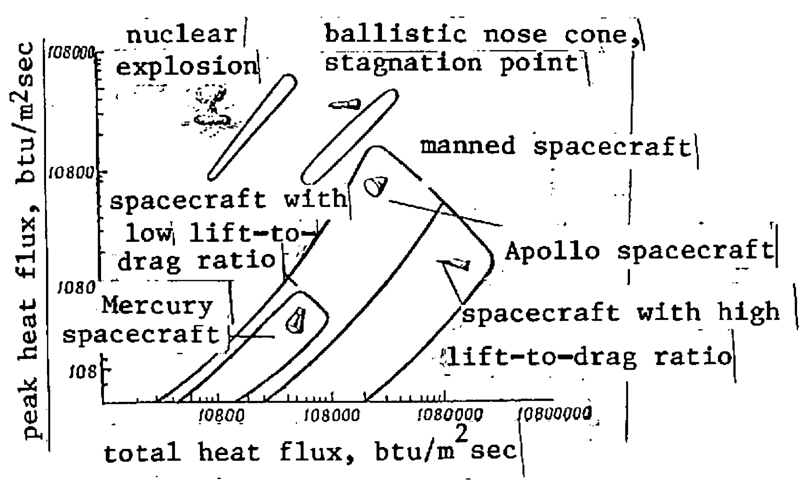

Figure 26. Thermal conditions typical of various spacecraft in atmospheric entry.

This method gives the so-called radiation heat protection system (Figure 27). Here the heat flux penetrating inside a spacecraft is usually small because of the efficient thermal insulation. By solving the heat balance equation at the operating temperatures of the blanket material, one can determine the maximum heat flux which the material is capable of taking. To create a radiation shield one requires a very thin layer of material, since its efficiency depends on the limiting allowable temperature and on the emissivity. The shield may be made of high-temperature materials such as molybdenum and nickel-steel with oxide coatings.

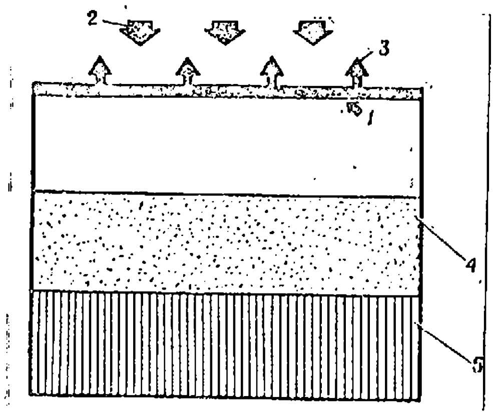

Figure 27. Radiant heat shield system: 1- external radiant surface; 2- incident heat flux; 3- reflected flux ; 4- thermal insulation layer; 5- supporting structure.

Another group of materials which can be used in radiation systems is refractory oxides (e.g., zirconium oxide) and ceramic materials. The maximum operating temperature of zirconium oxide is 2500°C, which allows it to be used for very large heat fluxes. However, it is difficult to use thin layers of these materials because of their brittleness. For graphite, well known as a material with high heat resistance, and which sublimes at a temperature of 3350°C, the maximum allowable heat flux is greater by a factor of 30 or 40 than for molybdenum alloys and steel. However, sometimes even this heat-resistant material is inadequate for protection against a very large heat flux.

Heat absorbing systems. The limitations associated with choosing materials for radiant systems can be overcome if a thick layer of thermal protection material is used. In this case the heat will not only be radiated from the material surface,

but will also be transmitted to inner layers of the material. These heat protection systems are called heat-absorbing systems (Figure 28).

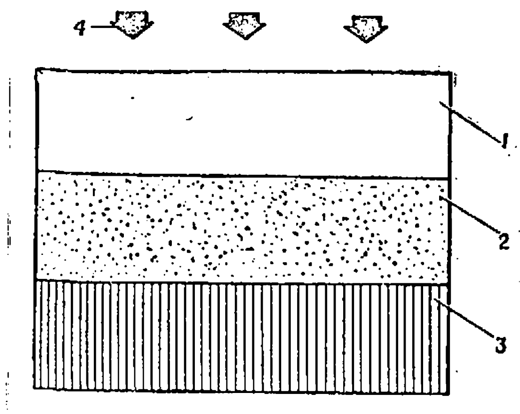

Figure 28. Heat shield system with absorbers.

1- heat-absorbing layer; 2- thermal insulation layer; 3- supporting structure; 4- incident heat flux.

In contrast with a radiant system, the heat is accumulated in the material in a heat absorbing system. There is a limit to the amount of heat which can be stored in a heat-absorbing layer of a specific configuration. This limit is the total amount of heat which can be absorbed before the material reaches such a high temperature that its strength decreases to an unacceptable level. Good heat absorbing materials must have high specific heat capacity, heat conduction, and melting temperature, in order to accumulate as much heat as possible before melting begins. Amongst the well- /72
known materials only copper and beryllium find use as heat absorbers, and were used on the nose sections of the first American ICBM's.

Ablation systems. A heat protection system with an absorber uses material with low heat conductivity. In this case the heat will not be removed from the surface of the heat protection layer to its inner layers. High temperature gradients will exist in the heat shield layer, and the surface temperature can exceed the melting temperature or the material breakdown temperature. But if the inner part of the heat shield layer remains quite cool and can withstand the stresses arising, then we can allow some of the material to be removed from the surface. This is the essence of the ablative heat shield. In this system the plan

is for some of the outer material of the heat shield layer to be removed in order to protect the inside. By ablation one means that the heat causes the removal of material, either by melting, sublimation, or some other process.

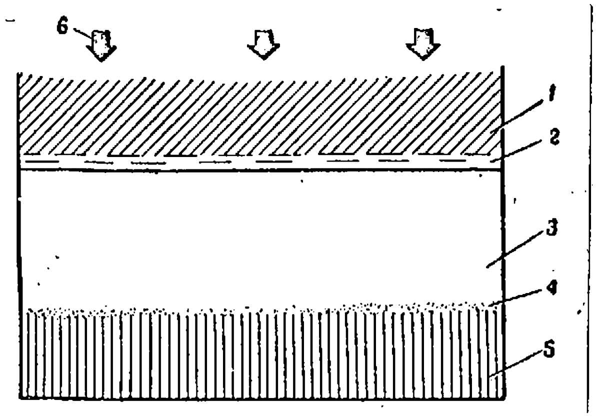

Figure 29. Ablative heat shield system.

1- char layer; 2- decomposition zone; 3- virgin ablative material; 4- bond; 5- support structure; 6- incident heat flux.

In comparison with other heat shield systems, ablation systems (Figure 29) are extremely efficient and withstand the action of high heat fluxes. The mechanism of ablation is very complex. We shall consider what happens if an element of a plastic surface begins to be heated at a high rate during atmospheric entry. When first heated the parts of the heat shield layer close to the surface acquire heat and in the main transmit the heat only by heat conduction. Finally the surface of the heat shield layer reaches the temperature at which pyrolysis of the resin begins. /73

As a result of pyrolysis, gaseous products and char or carbon solid residues are formed. If heating is continued, the pyrolysis front penetrates into the heat shield layer. The gaseous products now form at a certain depth in the layer and then begin to diffuse through the layer of char and are blown into the external stream. These ablation products can interact with the external flow in the boundary layer, and the combustion reaction becomes particularly important. In addition to reactions occurring in the gaseous medium, one can also have the phenomenon of recombination at the heat shield surface layer, when the heat flux reaches values high enough to produce dissociation in the free

stream. Besides combustion in the gaseous medium there can also be surface combustion whose intensity depends on the rate of the reaction taking place at low temperature of the char residue, or on the rate of diffusion of oxygen to the surface when the char residue temperature is high. If the spacecraft enters an atmosphere different from that of Earth (for example, into the atmosphere of Mars, which contains very little oxygen), then the efficiency of an ablative material can be increased markedly because of the absence of oxygen. This has been confirmed in engineering tests in an atmosphere of nitrogen and argon.

There have been many investigations whose objective is to strengthen a char layer and to attach it to the basic material. This kind of strengthening is often achieved by using a filler of silica or refractory oxides, whose use is limited by their melting at a certain depth.

In order to create a satisfactorily charring material one must endow it with specific properties. It must be material of the resin type, for example-phenol and epoxy resins. Quartz, glass and other fillers can be used to strengthen the char layer.

In order to protect the inside layer of the heat shield material from the effect of the temperature in the gasification zone, it is desirable that the material have low heat conduction. This can be achieved by reducing the density of the material, by using a honeycomb structure. Instead of reducing the heat conduction once can increase the breakdown temperature of the protective layer by introducing subliming inorganic salts. Two types of ablative materials should be mentioned. The first includes glass or fusible ablative materials, a typical member being quartz. If the heat flux is great enough to fuse and evaporate quartz, the tangential stresses due to friction of air in the boundary layer are not large enough to produce flow of the fused quartz

before it evaporates, and an efficient heat shield system can be achieved. The second type includes subliming ablative materials (for example, teflon). These materials do not form a char residue, and the gas is formed directly from the solid phase. The heat shield properties of ablative materials of this type are /74 not great, since their sublimation temperatures are low, and there is no loss of heat due to radiation from the surface of the heat shield. The mechanism of radiation is of primary importance for the ablation materials used in the heat shields of manned spacecraft. In the absence of a char residue, the mass of the heat shield system is increased several times. However, there is one advantage in favor of using non-charring ablative materials. The fact is that the char greatly attenuates the incoming radio signals, and therefore the antennas must be shielded by materials which do not form char residue.

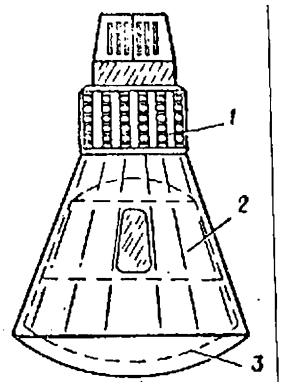

Figure 30. Heat shield system on the Mercury spacecraft: 1- heat-absorber; 2- radiator; 3- ablator.

Each of the three heat shield systems (radiant, heat absorbing, and ablative) has its area of usefulness. One of the American spacecraft used all three systems (Figure 30). The largest heat fluxes to its surface were observed in the blunted nose section, whose heat shield used an ablative covering, reinforced with fiberglass. To protect the conical section from the heat flux, a radiating coating was used — "tiles" of nickel-cobalt alloy. To protect the crew section, heat-absorbing tiles of beryllium plate were used.

Numerous combinations of the basic heat shield systems are possible. In particular, an ablative coating superimposed on a radiating layer forms a very good shield, if good bonding can be achieved. In this system the ablative material will act at

low heat loads, and the radiant layer will operate later, during atmospheric entry.

Another variant is an active efflux system, in which hydrogen or another coolant is blown through a porous plate of metal, ceramic material or graphite.

In considering the effect of heat it should be remembered that the heat shield layer is the outer layer of the crew compartment of a spacecraft entering the atmosphere. Therefore, it must withstand collisions with meteoritic particles, and also the action of ionizing radiation, ultraviolet light, and temperature extremes.

The functional performance of heat shield systems under these complex conditions must be examined, not only as a whole, but in the individual sections; for example — in sections where there are hatches, windows, gaps for engineering purposes, and apertures for nozzles.

CHAPTER 4

NAVIGATION AND GUIDANCE SYSTEMS

One of the most important functions of the navigation and guidance systems of a space vehicle is to correct the flight /75 trajectory. The basic difference between an orbital flight around the Earth and an interplanetary flight is the following: in a flight around the Earth a change in the orbit does not have a major effect on the possibility of a rapid atmospheric re-entry, but in an interplanetary flight, the success of the mission depends entirely on the accuracy of trajectory correction. In spite of the fact that the basic navigation and guidance systems are inertial, special attention must be given to the accuracy of operation of the guidance system during a correction.

For navigation, both in the Earth-Moon section and in an orbit around the Moon, a complex guidance system must be used. The system is redundant and is amplified by ground methods for accomplishing the interplanetary flight. In a flight in a geocentric orbit, the requirements for the navigation and guidance systems are significantly lower than for a flight around the Moon, and it is therefore considered sufficient to combine ground facilities and a single inertial system with a gyrostabilized platform relative to the horizon.

During rendezvous of two spacecraft, the initial section is accomplished by appropriate change in the orbit, while the next section, that of approach up to the distance of direct view of the spacecraft, is accomplished by means of onboard measuring devices. The final approach and docking is accomplished by manual control during direct visibility of the spacecraft. Manual control with visual observation can be accomplished without spe-

cial difficulty, and does not require any other guidance facilities. During a landing on the Moon, the attitude control is accomplished both automatically and by manual control when there is direct visibility of the Moon or the spacecraft. To assist the pilot, the spacecraft carries a radar equipment and some other measuring devices. During the return to Earth, the guidance in the atmospheric re-entry uses the aerodynamic force created by maintaining a constant angle of attack of the spacecraft. Guidance signals to ensure the required trajectory are processed by a computer on the basis of information arriving from the inertial system. /76

If the basic guidance system malfunctions, simple redundant systems are brought into operation.

The navigation and guidance systems include systems for time measurement, and inertial, optical, and radar systems, systems for attitude control and stabilization, and reactive control systems.

The time measurement system is used to fix the time of all processes and operations occurring and being recorded on the spacecraft. The presence of computers in some of the systems makes it necessary to use a relatively high frequency time measurement system, capable of high accuracy. Since a computer determines many of the processes, its operation must be accomplished from a self-synchronizing device operating synchronously with the time measurement system. A redundant time measurement system must either operate continuously, in parallel with the main system, or be switched on only after the basic system becomes unserviceable. In both cases synchronization is required. When different kinds of reserve systems are used simultaneously, ordinary mechanical clocks are employed, which can determine the given time for the majority of processes, and can be useful for

monitoring the system. Good results can be obtained by a combination of mechanical and electronic clocks, but the problem of synchronizing them is then complicated appreciably.

Inertial systems are used for measurement and integration (summing) of acceleration in the determination of spacecraft position. A gyro-stabilizing device must be used in this kind of system to give the coordinate system.

At present, a factor governing the reliability of an inertial system is the wear of the gyroscope bearings. Successful development of inertial systems for new equipment can alter the nature of their construction. Gyroscopes with air bearings and gyroscopes with electrical supports are very promising in this respect. There has also been improvement in integrating accelerometers, used in many inertial navigation systems.

<u>Optical systems</u> are used to determine the spacecraft position in an inertial coordinate system. The horizon sensor detects the local vertical by determining the direction to the edge of the visible disc of the planet. These sensors compare the levels of infrared rays coming from the Earth with the almost zero level of radiation from deep space. The difference in the radiation level of infrared rays is independent of the time of day or night and causes a sharp drop in radiation at the horizon. This drop is observed at the edges of clouds at the interface between day and night. This radiation drop can be amplified by the method of frequency selection, based on the wavelength difference between reflected solar radiation and radiation from the Earth's troposphere. This method gives quite efficient resolution of the useful signal in most cases, apart from the period of sunrise and sunset.

One of the optical instruments used in space flight is the

sextant. It is used for navigation in interplanetary flight (including flight to the Moon), when it is necessary to determine accurately the position of the spacecraft in space. It is a complex technical matter to choose the line of sight to the Moon or another planet before beginning navigation in the immediate vicinity of a planet. This is the very reason that one can employ the capability of the human being to determine orientation for line of sight comparatively simply. The capability of the human to orient himself by the stars also makes it possible to simplify the system of observation and target designation employed in automatic methods. It is clear that a manual system is much simpler and more reliable. In addition, a human being can orient himself by stars whose brightness is several orders of magnitude less than that required for automatic systems.

Radar systems are used as measuring devices for navigation systems onboard a space vehicle and on the earth.

Ground-based radar stations offer high accuracy of measurement of range and rate of change of range. Their accuracy in determining angular values is quite satisfactory, but decreases as a vehicle becomes more remote from the Earth. The region of operation of radar stations on Earth and in lunar orbit is restricted to direct visibility, but autonomous navigation is always possible, and the accuracy of measurement increases as the distance between the space vehicle and the object to be measured decreases. Thus, it is reasonable to consider combined systems in which ground and onboard radar facilities are used in conjunction.

To obtain a strong enough return signal, a spaceship carries an onboard radar transponder. Onboard radar stations are used during maneuvers to achieve rendezvous in orbit, to measure the

range and rate of change of range to the target vehicle.

During a landing on the lunar surface, the radar instrument is used as the principal measuring element of the navigation system. The basic system is an inertial navigation system which immediately corrects the results of measurement of distance to the lunar surface prior to the final descent, in order to achieve the required accuracy during a landing on the surface. The landing is accomplished, if possible, using visual observation and a three-path Doppler altimeter. A radar instrument can also be used on the lunar excursion module for rendezcous in a selenocentric orbit.

A computer performs calculations and supplies the signals /78
for the motors which create the control moments. The computer input receives signals from the measuring elements which determine the angle of orientation and the angular rates of the spacecraft, as well as information on the required maneuver or attitude change. In certain cases, to reduce the usage of fuel, the computer only applies limits to the angular rate of rotation of a spacecraft at a specific level. In many cases the purpose of the control motors is to produce translational and rotational motion of the spacecraft. The computer has a logical scheme for choosing a control motor, to determine which motors should be used to accomplish the required movement. When there is to be simultaneous translational and rotational motion it may happen, for example, that the translational motion motor will operate in a direction opposing that of the rotational motion motor. Then no motion results, and there will only be a useless expenditure of fuel.

The choice of the control motor logic system eliminates such operating modes by blocking the signal to switch on one or all of the motors. The computer on the Apollo spacecraft had a con-

trol mode which only sought to stabilize the angular rate and was used while the spacecraft motion was being controlled. In the attitude control mode, the spacecraft was stabilized in a given attitude, or a change of operation was effected upon command of the astronaut or of the computer.

In the control mode, the local vertical is used to effect attitude control of the spacecraft with respect to the vertical on the Moon or the Earth. In the atmospheric re-entry mode the functions are: attitude control immediately before atmospheric entry, stabilization in pitch and heading during flight in the atmosphere, and rapid changes in roll angle which are required to control the aeordynamic lift in the atmospheric flight section.

A very simple system for developing control commands for the engine is used in the control computer, and operates as follows. An error signal, consisting of spacecraft speed and angular position signals, is input to a switching amplifier. As long as the error signal does not go outside the limits of a given dead zone, there is no signal at the amplifier output. When the input error signal goes beyond the dead region, the amplifier creates a "start" signal and the control motor begins to operate. The start signal remains at the amplifier output until the error signal again falls within the dead band. /79

The reliability of onboard computers can be increased by replacing elements that go unserviceable immediately in the flight, but to do this one must carry out a complicated check of the computer (to discover the malfunction), using a special unit and with the astronaut taking part, or else use a triple reserve system with automatic detection of malfunctions and replacement of the defective elements.

Attitude control and stabilization systems. One of the main

requirements for the control system is to establish and maintain the required angular position of the spacecraft.

The requirements imposed on the control law differ both for different flight missions, and in different sections of a spacecraft flight. In space flight of an uncontrolled spacecraft the damping and restoring moments are zero. Therefore, any perturbation acting on the spacecraft causes it to rotate. It will continue until an opposing rotational moment of opposite direction is applied to the spacecraft. In order to compensate even for very small external or internal perturbing moments, one must have a system for attitude control of the spacecraft. The restoring moments of the attitude control systems are accomplished with the expenditure of a very small mass of fuel even during a long flight of the spacecraft. The device for creating control moments is the low-thrust rocket motor.

The maximum control moment is determined by the necessary angular acceleration for a maneuver and the maximum possible perturbing moment. In many cases the maximum perturbing moment arises when the main engine of the spacecraft is operating, where the thrust vector of the motor does not coincide with the center of mass of the spacecraft. The control system must offer maneuverability, attitude control of the vector thrust, and appropriate accuracy in control when the velocity is changing. The communication and electrical systems require a specific orientation of the antenna or of the solar cells. A specific orientation of the spacecraft relative to the Sun may be required for reasons of thermal control.

In accordance with these requirements, the control system can be manual, semi-automatic or automatic. Experience shows that manual control (Figure 31) can be quite efficient and that it is desirable to enable the astronaut to go to automatic con-

trol when there is a need for precise control which he cannot insure on his own.

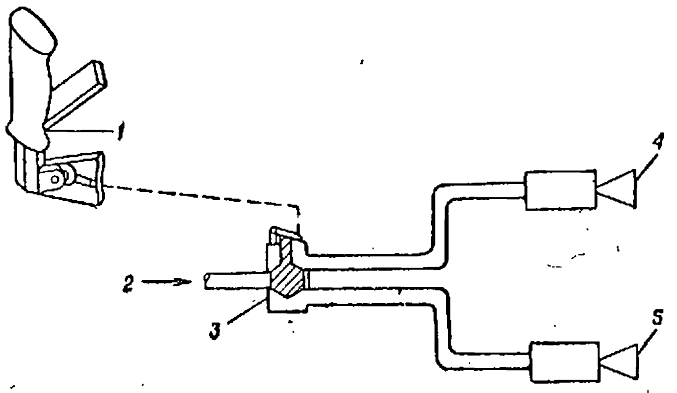

Figure 31. Manual single-channel proportional control system: 1- manual control; 2- fuel supply; 3- throttle valve; 4- thrust in the positive direction; 5- thrust in the negative direction.

The manual control system consists of a handle which deflects in three directions and is mechanically coupled to valves. The valves regulate the supply of fuel to create motor thrust and angular accelerations proportional to the amount of deflection of the control stick. A clear advantage of this system is that /80 it is totally independent of electrical energy and it can compensate for long-duration perturbing moments, while a disadvantage is the complex kinematic control of the valves, which increases the mass of the spacecraft. In addition, the manual control does not experience a force proportional to the engine thrust, and the astronaut does not sense the moment which he has applied.

In a wire control system (Figure 32) three-axis manual control is used, with angular deviations which operate limit switches, so that a voltage from an onboard source is supplied directly to the winding of the control motor solenoid valve. For a small /81 smooth deviation of the stick, the spacecraft receives a velocity increment, but not a constant acceleration, which is obtained for normal deviation of the stick. It has been established that the minimum duration of a pulse which the astronaut can give with

the manual control is roughly 100 msec. This pulse duration can give velocity increments which are too large for special operations (e.g., for docking in orbit or for navigational determination of the spacecraft position and coordinates). Therefore, direct control of the engine is applied, particularly in the emergency regime, and for navigational determination of the spacecraft position and coordinates, and additional elements are used to achieve smooth control.

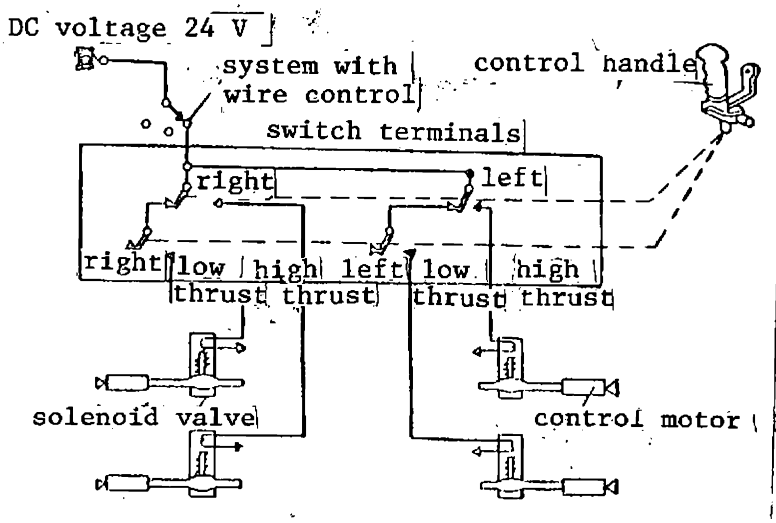

Figure 32. System with wire controls.

In the angular rate control, as in the previous system, a command is given manually, but the system provides damping, and can therefore be classified as a semi-automatic control and stabilization system.

In an automatic stabilization system, each of the control channels is a closed automatic control system, consisting of a measuring device, an electronic control scheme, an auxiliary mechanism, and a feedback circuit for damping spacecraft oscillations, and also to provide information concerning the required maneuver or orientation.

The operation of the main engine which corrects the space-

craft trajectory in the cruise section is accompanied by considerable perturbing moments when the thrust vector does not pass through the spacecraft center of mass. When the thrust is large and the duration of action of the main motor is long, a control system to compensate for the perturbing moment must create large control moments. For this reason the main engine is usually mounted on gimbals.

Control of the motor attitude is accomplished by an automatic position control system which uses error signals from sensing elements of the control system. The motor mounted on gimbals maintains attitude control only in heading and pitch. Roll stabilization during operation of the main engine must be achieved by a jet control system.

Control system jets. These systems are used on a spacecraft in the absence of aerodynamic forces to perform maneuvers and control the spacecraft. They change a spacecraft attitude in space or impart a rotary motion to it by means of thrust impulses or by changing the angular momentum of the vehicle.

All the systems used to control a spacecraft can be divided into three groups: nozzle jets which create a thrust by expanding a mass; solar sails or magnetic correction motors, which create a control force by interaction with the surrounding fields; and devices to transmit angular momentum which do not create control /82
force, but simply transmit angular momentum to or from the vehicle.

Use of a solar sail or a magnetic correction motor is limited to equipment which requires very small control forces.

Manned space vehicles require comparatively large control forces, which are usually created by nozzle jets. A control

system with nozzle jets differs from other spacecraft force systems as regards the thrust, the mode of operation, and the possibility of repeated use. The maximum thrust is determined by the magnitude of the perturbations which must be cancelled, or by the magnitudes of the angular or translational accelerations required to accomplish the various maneuvers. Control systems use constant thrust, and the duration of operation for automatic systems varies from 3 to 5 millisecond up to several minutes per pulse. Manual control systems use motors with variable thrust. Usually the thrust of the motors of a jet control system is considerably less than that of other spacecraft force systems. It is stated in the foreign literature that the nominal thrust of jet system motors for control of the crew module and the equipment module* of the Apollo spacecraft was about 45 kg.sec, while the thrust of the main power unit of the equipment module was more than 9000 kg.sec.

The mode of operation of constant thrust motors is typically that of variable duration, together with variable frequency of pulse repetition. The pulse repetition frequency can vary from several pulses per second to a single pulse in several minutes.

The working substance for control jet systems include: compressed gas, and mono-component and binary-component fuel.

A compressed gas control system is simple and very reliable. However, the density of gaseous fuel for storage is low. As the required thrust pulse increases, the size of the storage tank must increase. Therefore, this type of system is used when the energy required is low.

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* The equipment module, the service module, the auxiliary module, and the engine module (EM) used identical concepts.

It is proposed that control jets for a spacecraft on a flight to Mars can use hydrogen vapor, vaporized in the fuel tank of the main power installation. When heated in a nuclear reactor to very high temperature, these vapors can have a specific impulse of the order of 700 sec. In this case the control jet system does not need a subsystem for fuel storage. One of /83 the chief problems if a heat vapor system is used is the heat source configuration.

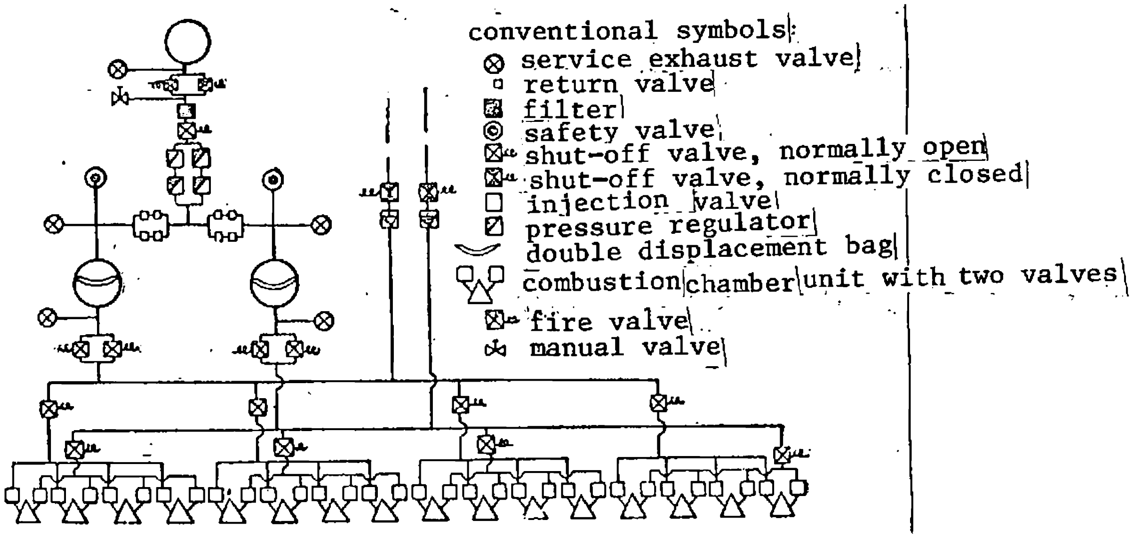

Figure 33. Duo-component fuel control system with reserve elements.

A mono-component propellant control system uses a single working fluid, which, expanding in a nozzle, undergoes chemical and thermodynamic processes in the presence of a catalyst, to create a thrust. A frequently used fuel of this type is highly concentrated hydrogen peroxide, which, in decomposing, forms superheated water vapor and oxygen gas. Hydrazine is also used as a mono-component propellant. The principal advantages of a mono-component propellant system include comparative simplicity, high reliability, and low gas temperature. The system has only one fuel tank, one feedline system, and one valve. This, together with the good reliability, makes a mono-component propellant system preferable up to a certain impulse level. However, as

flight missions become more complex, the mass of the combustion chamber, the feedline system and the valves become a smaller part of the total system mass and it is therefore more suitable to use a duo-component propellant system which possesses better performance characteristics.

In duo-component propellant systems (Figure 33) the fuel is usually a substance of the hydrazine family, while the oxidizer is nitrogen tetroxide. This duo-component propellant creates a significantly larger impulse in comparison with a mono-component propellant, which may be increased by using metallic hydrides as a high-calorie fuel. The main element of a control jet system is /84 the motor with its combustion chamber, cooled by radiation or ablation. The chamber walls are made very thin so that the temperature of the inside chamber wall, which is cooled by radiation, should not become excessively high. The materials used for the chamber are heat-resistant metals, such as niobium, molybdenum, tantalum, and tungsten, or their alloys. The foreign literature states that the engines for the equipment module of the Apollo spacecraft and for the lunar excursion module were radiation cooled and made of molybdenum alloy with titanium additive. Ablatively cooled chamber walls are made of heat-resistant materials (phenol and epoxy resins). When the combustion chamber is ablatively cooled, pyrolysis of the binder occurs. The heat supplied to the inside walls of the combustion chamber is absorbed by pyrolysis of the binder, and at the upper non-irradiated part of the material a porous base of charred binder and heat-resistant material is formed. The gaseous pyrolysis products return through the charred material and pass into the boundary layer at the combustion chamber wall. In this way they shield the chamber walls from the additional heat coming from the hot combustion products. During operation of the motor, the interface between the charred and uncharred material approaches the inside surface of the combustion chamber. When the interface reaches the

inside chamber wall, the engine ceases to function. Therefore, an ablatively cooled engine has a limited service life, depending on the initial thickness of the combustion chamber wall. Periodic switching on and off of the engine presents a more severe regime than continuous operation, since the hot charred surface continues to transmit heat from the remaining mass of material between the pulses.

The ablative material is a good insulator, and therefore the inner surface of the motor remains relatively cool during operation. Consequently, an ablatively cooled combustion chamber can be located inside the body of a re-entering spacecraft where it will not be exposed to aerodynamic heating and aerodynamic forces.

The charred ablation material is relatively soft. At the nozzle throat it cannot withstand the high-speed flow and tangential stresses, and therefore, material is carried away at this location, and the area of the throat section gradually increases. To avoid removal of mass one must use strong heat-resistant coatings in the nozzle to keep the throat cross-sectional area constant during the entire lifetime of the chamber. /85

A complex of control system jet engines also includes an injection system with an atomizer, which injects the fuel and oxidizer in the required ratio into the combustion chamber.

In choosing a spacecraft system, a good deal of importance is attached to the reliability of the attitude control systems. Increased reliability of the control jet system, as is true for any other complex system, is attained by simplifying the arrangements, and especially by reducing the number of sequentially operating elements. However, if reserve elements can be used in the system (then a malfunction of one element does not put the system out of action), the reliability of the system is increased

appreciably. In designing control jet systems for the American spacecraft, use was made both of individual spare elements and also duplication of the entire system. In particular, the equipment module of the Apollo spacecraft has four independent systems, any two of which are capable of accomplishing a safe return of the equipment module, while the crew module has two completely duplicated systems.

CHAPTER 5

SPACE VEHICLE ENGINES

/86

We now consider the sequence of flight stages of a space vehicle around the Earth and the operation of the engine complex at this time.

A contemporary space vehicle is injected from Earth into an orbit with the necessary speed, by means of a launch-vehicle. After the launch-vehicle separates from a spacecraft, the onboard engines can be used to accomplish maneuvers of the spacecraft in a steady orbit or to transfer it to another orbit around the Earth. When it is necessary to stop the flight, the onboard engine is used to decrease the spacecraft flight velocity, and the spacecraft re-enters the atmosphere, where the flight speed is reduced because of aerodynamic drag. During atmospheric re-entry one must control the spacecraft attitude-control system and maintain the flight trajectory, to accomplish a landing at the required location. The engine unit whose function is to maintain spacecraft attitude is called the control jet system. The maneuverability of a spacecraft in orbit must increase as the problems of orbital flight become more complex (rendezvous and docking of two spacecraft, direct observation of other orbiting spacecraft, and flight over specific regions of the Earth's surface).

For the flight of a spacecraft involving a lunar landing, repeated operation of the power unit is required (Figure 34). In particular, after injection of the Apollo spacecraft by its launch vehicle into a flight trajectory to the Moon, the first of several active maneuvers was performed, a so-called circular turn,

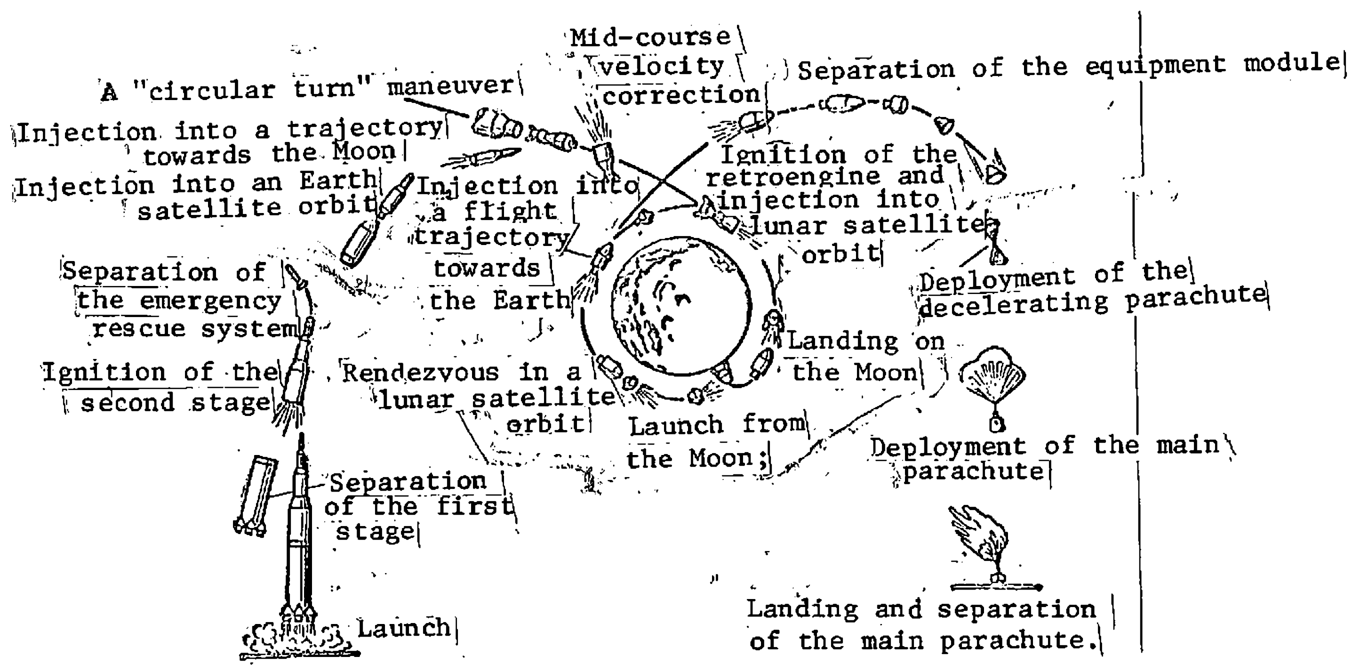

Figure 34. Flight sections for a spacecraft with a landing on the Moon.

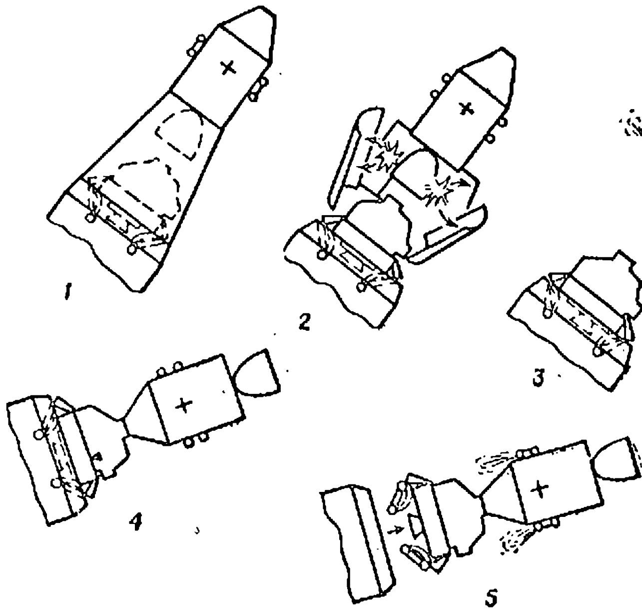

Figure 35. Various configurations of the Apollo spacecraft.
1- injection into a trajectory towards the Moon; 2- Separation of the transfer module; 3- freely tumbling; 4- docking in the backward position; 5- separation of the spacecraft from the Saturn IVB stage.

in which the four engines of the control jet system located in the forward part of the spacecraft were switched on for a short time and separated the crew and main instrument modules from the intermediate stage and the spent launch-vehicle (Figure 35). To achieve the required distance between them, the engines mounted on the rear part of the spacecraft were operated, reducing the relative velocity to zero. Then, after a certain time, the engines located in the rear and forward parts of the spacecraft were switched on, and created a moment causing rotation. After /88
the spacecraft had rolled through 180 deg, these engines created an opposing moment which stopped the rotation. Then the engines located in the rear part of the spacecraft were brought on again, to cause the spacecraft to close with the module. Here the speed and attitude of the spacecraft were

controlled by appropriate engines until there was a smooth mating with the lunar excursion module. After the first docking maneuver was accomplished, the composite spacecraft was separated from the launch-vehicle by periodic burns of the four motors located in the forward part of the spacecraft.

A space vehicle of the Apollo type is equipped with seven different engine systems, each of which is required for a successful mission and performs different functions. What are the special features of the technical characteristics of the onboard engine system of the spacecraft and what are its requirements?

Above all there is the need for the system to be adapted to the operational conditions (vibration and noise during launch, aerodynamic loads and heating, high vacuum, the weightless state, a wide range of temperature, from the high temperatures resulting from heating by solar radiation to the low temperatures due to loss of heat by radiation to deep space, a radiative flux, the effect of corrosive fuels, etc.).

The spacecraft engine system can be divided into the propellant supply system, which stores the propellant and supplies the necessary quantity to the engine, and the rocket engine, in which the chemical energy of the propellant is converted into a thrust of the required magnitude. One of the most important parameters describing the performance of an engine is the specific impulse, i.e., the thrust developed per unit mass of propellant. The mass flow rate of propellant depends on the engine geometry, the total pressure in the combustion chamber, the temperature of the combustion products, and several other factors.

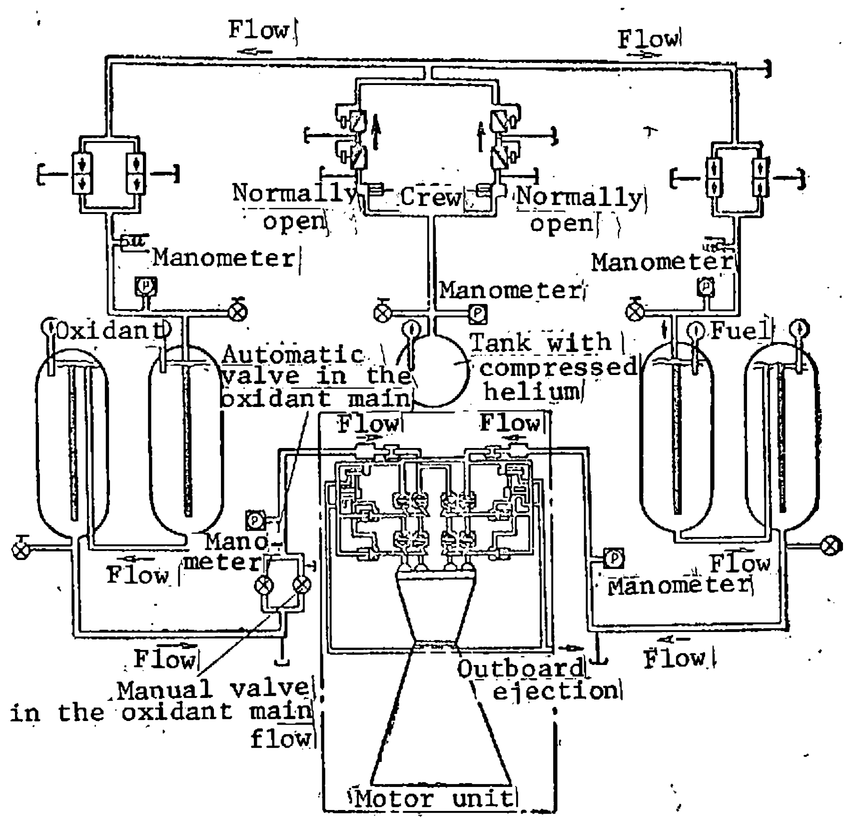

Figure 36. Schematic of the motor system of the equipment module of the Apollo spacecraft (the arrows and the word "flow" indicate the direction of the flow of gas and propellant components).

For docking of spacecraft, any errors in speed must be corrected by changing the engine thrust, before direct contact takes place. If this is not done then either docking is not achieved, or the spacecraft collide. Since the range of observation of spacecraft is limited by the capability of the onboard radar and other navigation devices, careful control must be exercised over the relative speed of approach. The engine thrust must carefully reduce to zero the residual errors in the speed at the distance of the final approach to the target. The problem is complicated by the fact that each of the docking vehicles or modules can have six velocity components: three translational and three rotational. To center the docking devices, one needs a

specific mutual position of the spacecraft, and, before docking occurs, all six of the relative velocities must be reduced to zero. /89

The need to control spacecraft velocity during descent to the lunar surface and landing also imposes specific requirements on the engine system. For a successful mission, the spacecraft must be capable of cancelling its velocity, rapidly gaining and losing height, and moving horizontally over the lunar surface. Since these maneuvers required a considerable amount of propellant, the time that they take must be reduced to a minimum.

The equipment module of the Apollo spacecraft has a control jet system (Figure 36) which consists of sixteen engines, located in four groups of four uniformly around the perimeter of the spacecraft. This positioning of the engines permits translational movement along all three axes, and in addition, allows rotation relative to these axes if only two motors in each group are switched on. /90

During attitude control of the spacecraft or an accurate correction, the flow rate of fuel depends considerably on the method of accomplishing these operations. When an engine operates, acceleration is created. If an engine with a short period of operation is used to give a spacecraft a small velocity and subsequently it is cancelled, then a small amount of fuel is used. However, if an engine operates during the entire time for a correction, then the correction is accomplished in the minimum time at the maximum rate, which leads to a maximum expenditure of fuel. The onboard spacecraft engine unit must be large enough to accomplish a change in velocity over the complete range. For flight of a spacecraft to the Moon, the change in velocity is several thousand meters per second, and the motor must develop a thrust of about 10,000 kg/sec. However, to accomplish accurate navigation and docking, the velocity of a spacecraft must be

controlled to a few meters per second. For such an accurate velocity correction, one can use the engines of the control jet system, but their thrust must be small in order to generate a minimum impulse. Thus, to vary the velocity over the complete range, from several thousands to tenths of a meter per second, engines are required with a long operating time and small fuel stock, as well as a low-thrust engine capable of giving a minimum impulse.

The choice of the engine unit for a spacecraft is quite a complex matter. There is a multitude of possible combinations of fuel components, structure of fuel tanks, supply systems, and many other assemblies.

At present liquid rocket engines are used for the main engines of manned spacecraft. High-thrust solid fuel engines, nuclear power units, electrical rockets and engines of other types and arrangements have very limited application in contemporary spacecraft.

LIQUID ROCKET ENGINES (LRE)

A schematic diagram of a liquid rocket engine is shown in Figure 37. The liquid fuel and the liquid oxidant are located in separate tanks and are continuously fed by the supply system into the combustion chamber where the mixture burns. The discharge of gas from the rocket motor nozzle arises exclusively from combustion of substances located on the vehicle itself. This circumstance makes the operation of the rocket engine independent of the surrounding atmosphere and is the main reason for the use of these engines in launch-vehicles and in spacecraft.

A liquid rocket engine was first proposed in 1903 by the eminent Russian scientist and inventor, K. E. Tsiolkovskii, who

published his paper "Investigation of Outer Space by Means of Jet Devices" in the journal "Obozrenie" (The Inventor). The paper /91 gave a detailed description of a rocket device and its liquid rocket engine. The scheme for the engine and the principle for some of the solutions proposed by Tsiolkovskii remain unchanged even for present-day liquid rocket engines.

In liquid rocket engines the propellant consists of an oxidant and a fuel, which are usually called the propellant components. Burning of the propellant and expansion of the combustion products occurs in the combustion chamber and the jet nozzle of the engine.

The fuel and oxidant are supplied to the combustion chamber through the fuel jets, located at the head of the chamber. The propellant is supplied to the combustion chamber at a pressure greater than that inside the chamber. The supply of propellant is accomplished by means of a special system (the supply system), which consists of an assembly to create pressure (tank of compressed air or turbo-pump unit), as well as pipes with valves and regulators.

The thermodynamic process in a liquid rocket engine is substantially different from that occurring in jet engines of other types. The propellant must arrive in the combustion chamber in a state favorable for the combustion reaction. This means that the fuel and the oxidant must form as homogeneous a mixture as possible. To do this, an atomizer is used to spray them as very fine drops and to mix them within the required proportion in the combustion chamber. Entering into a mutual chemical action, the propellant components ignite and burn. When the propellant burns, its stored chemical energy is converted to some extent into heat, increasing the heat content of the combustion products. /92
The work done in the expansion increases the speed of the

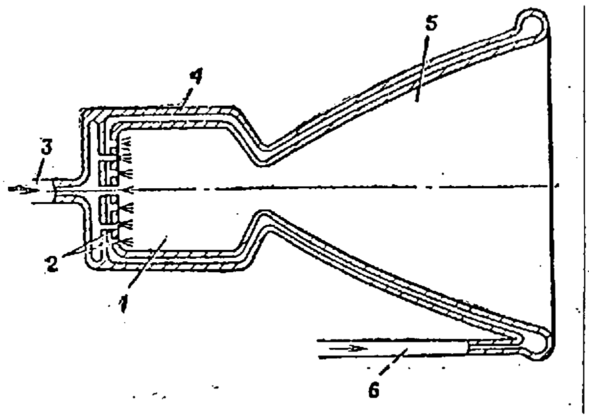

Figure 37. Schematic of a liquid rocket engine.
1- combustion chamber; 2- atomizers; 3- oxidant supply tubes; 4- cooling jacket; 5- nozzle; 6- fuel supply tube.

combustion products discharging from the nozzle, and therefore, creates thrust. The thrust force of the engine depends on the amount of gas discharging from the engine, and its speed of discharge relative to the motor. The combustion products give up heat very vigorously to the walls of the combustion chamber and nozzle. The combustion chamber is heated vigorously and must be cooled. A cooling sleeve is used for this purpose. One of the components, usually the fuel, passes prior to reaching the combustion chamber through channels, which form the chamber wall and a sleeve, and give up heat to the walls.

Liquid rocket engines differ as regards the type of propellant used, the method of supplying the propellant components, and the function of the motor.

Liquid rocket engines use propellants of two basic types: unitary and separate supply. A unitary liquid propellant is a single substance (or a solution of several substances), in a form ready for combustion (or decomposition). This type of

propellant could be called liquid gun powder. A separate supply propellant consists of a fuel and an oxidant. It is supplied to the combustion chamber separately and is mixed only in the actual chamber.

The processes of atomization and mixture-formation in engines using these propellants differ considerably. In engines operating with unitary propellants, the components taking part in the combustion reaction are mixed beforehand, and in the atomization process, it is necessary only to distribute the propellant uniformly over the combustion chamber cross section. In separate supply engines, the atomization process must achieve careful mixing of the fuel and oxidant particles to create the optimum conditions for combustion, and consequently for the most complete possible release of the chemical energy of the propellant. In engines operating with unitary propellants, the simplest supply system is to use one tank and one feedline between the tank and the combustion chamber. However, it is difficult to create a unitary propellant because of the explosion hazard.

Separate supply engines are further classified according to type of oxidant used. The properties of the oxidant determine the structural design of the engine to a considerable extent, and very often determine the possible use of the engine in a specific spacecraft. Engines are usually classified according to type of oxidant used: oxygen nitrous oxide, etc. At the present time oxygen and nitrous oxide engines are the most used.

The supply of the propellant components to the combustion chamber of a liquid rocket engine is accomplished mainly by two methods: discharge of the components from tanks by creating an excess pressure in the tank (the displacement method), or supply of the components by means of pumps. /93

The tank pressure for discharge can be provided in different ways. A supply system is widely used where the components are discharged by means of a high pressure gas (air, nitrogen or helium). This system operates as follows (Figure 38). A gas tank with a reducer to decrease the pressure to the working level in the tanks is fed into the tank with the propellant components. The gas pressure causes the propellant component to discharge into the combustion chamber.

Liquid rocket engines also use a pumping feed system, for the components (Figure 39). These very often use a turbine to drive the pumps. The turbine and the pump are located in a unit called a turbo-pump. The pressure created by the pumps must exceed the combustion chamber pressure by roughly 10-20 atm (depending on the construction of the atomizers and the flow resistance in the cooling system). The capacity of the turbine which operates the pump is 5-8 l.sec. per kilogram of the liquid mass pumped per second.

The working substance used to drive the turbine is provided in different ways. One method is to obtain a working substance (vapor) by decomposing concentrated hydrogen peroxide or some other unitary propellant. This method is self-contained and provides quite a stable and reliable turbine drive. The vapor temperature from decomposition of hydrogen peroxide is usually 400-700 C.

A defect of this pump type propellant supply system is that, besides the main propellant components required to operate the liquid rocket engine, one must have in addition a special fuel to operate the turbine. To eliminate this defect, the working substance to drive the turbine in a number of liquid /94
rocket engine designs is obtained from the basic components of the engine fuel. To do this, the products of combustion of solid fuel can also be used.

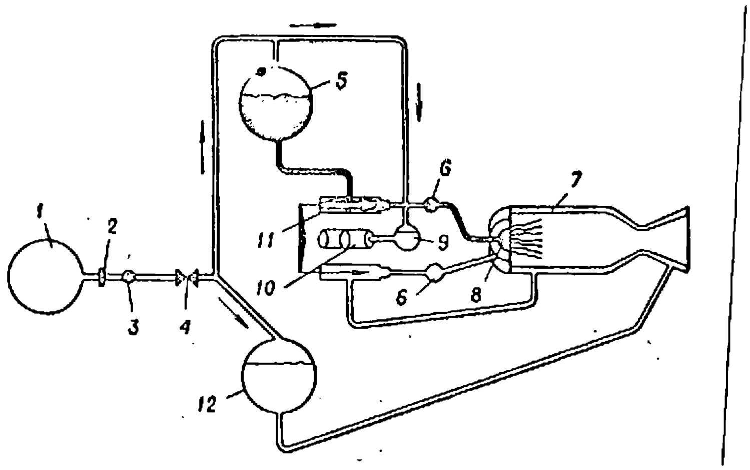

Figure 38. Displacement system for propellant supply in a liquid rocket motor.

1- gas tank; 2- tank valve; 3- gas pressure reducer; 4- motor start valve; 5- tank with fuel; 6- control heads; 7- combustion chamber; 8- atomizers; 9- hydraulic accumulator; 10- power cylinder; 11- propellant valves; 12- oxidant tank.

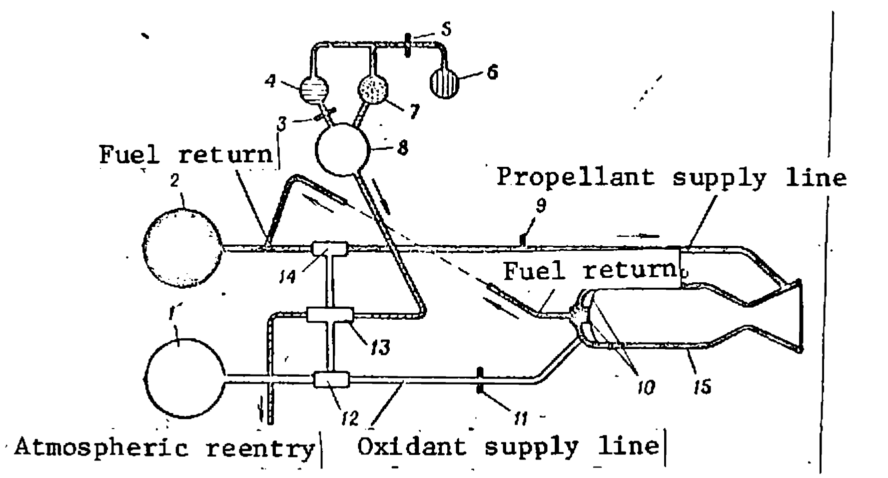

Figure 39. Pumping system for propellant supply in a liquid rocket motor.

1- oxidant tank; 2- fuel tank; 3,5- valve; 4- volume where the chemical composition is analyzed; 6- compressed gas tank; 7- volume for catalyst; 8- vapor-generator; 9- oxidant restart valve; 10- fuel and oxidant atomizers; 11- oxidant restart valve; 12- oxidant supply pump; 13- turbine; 14- fuel supply pump; 15- combustion chamber.

The name liquid rocket engine is given to both single-shot and multiple-pulse engines, used in various rockets and guided missiles, and to multiple-use and multiple-start engines which are used on space vehicles.

The most important stages of a liquid rocket engine operation are the start and the stop.

The start system must provide rapid and reliable ignition of the first charge of fuel and should not allow a significant increase in pressure in the combustion chamber during the start. Even a small delay in ignition leads to a considerable amount of fuel in the chamber, containing a large amount of stored energy. Rapid release of this energy can lead to an explosion of the fuel components and to disintegration of the spacecraft engine unit and even of the spacecraft. The start process is usually automatic. An automatic system provides the required sequence of all the operations at the time of start (e.g., start of the gas-generator to provide the vapor, start of the turbopump unit, switching on of the igniter, etc.). In the event of some kind of irregularity in the start system, the automatic control must shut down the start. Thus, if the ignition /95
does not function, the control system does not allow the fuel components to reach the combustion chamber.

The ignition of a liquid rocket engine during a start is accomplished in three ways: pyrotechnically, chemically, and electrically.

Pyrotechnic ignition is achieved by means of a special pyrotechnic device which turns for a very brief time interval, forming a high temperature jet (2000 C). This jet ignites the fuel supplied to the chamber. The pyrotechnic device is mounted in the head of the chamber or is inserted into the chamber from

the nozzle side on a special vane-holder. The holder has three devices arranged symmetrically around it. When the devices ignite, the vane begins to rotate, the chamber is filled with high temperature gases, and then a small amount of fuel is injected and burns. Then the amount of fuel supplied is increased. An incandescent electric wire is ordinarily used to ignite the pyrotechnic devices. Pyrotechnic ignition is used in engines which give a thrust of various amounts, in single-shot and multi-use engines, but with a single start.

Chemical ignition is used in engines which operate with non-self-igniting materials and develop thrust of any value, and are single-start or multi-start types. In this case, a special ignition system is provided in the LRE supply, with self-ignition components. To start the LRE, these components are first injected into a combustion chamber, are self-ignited, and form a flame site. The main fuel components are supplied to the combustion chamber thereafter. This engine is less hazardous in operation and is cheaper than an LRE operating only with self-ignition components.

Electrical ignition (using a spark plug) is used in low-thrust engines and in small experimental engines, intended for engineering tests. The defect of this means of ignition is the comparatively small heating power of the plug. In addition, an electrical energy source is required for electrical ignition, and this cannot always be provided on a flight vehicle.

An LRE can be stopped either by complete depletion of the fuel from the tanks, or by closing the valves which were opened to admit the fuel components into the combustion chamber. These valves are called cut-off valves. After the engine is stopped by closing the cut-off valves, a small amount of fuel reaches the combustion chamber from the pipes between the cut-off valve and the jets. For operational safety, this fuel is vented from the chamber by means of compressed gas.

Liquid rocket motors, as has been mentioned, can operate at any height in the atmosphere and in airless space. They provide a large thrust for a small mass and size. /96

Contemporary LRE's are used for ballistic rockets, high-altitude rockets, and spacecraft. They are also used in air-to-ground rockets and in air-to-air rockets.

The construction of LRE's is largely determined by their function. They can be both single-chamber or multi-chamber types.

To impart the required speed to spacecraft and ballistic rockets, a large amount of energy must be expended, and this is obtained by combustion of a large amount of fuel. To acquire high speed, a launch vehicle must carry a large stock of fuel and therefore, the initial or launch mass must be large. Staging is used to reduce the launch mass of a launch vehicle. Each stage has its own LRE and fuel tank.

In launching a two-stage rocket, the first-stage engine is ignited first, and this accelerates the rocket to a certain speed. After the first-stage fuel is depleted, the first-stage separates from the rocket, and second-stage engine is ignited, and accelerates the rocket to its final speed. Thus, only the second stage is accelerated to the final speed in this rocket, whereas the whole rocket is accelerated in the first-stage burn. Therefore, the expenditure of fuel to give the payload the final speed in a two-stage rocket is less than in any single-stage rocket. Therefore, for the same payload and flight range, the initial mass of a multi-stage rocket is less than that of a single-stage rocket.

In particular, the Apollo spacecraft uses the three-stage Saturn launch vehicle. The first stage is equipped with five LRE's with a thrust of 690 tons each. The first-stage oxidant is oxygen, and the fuel is kerosene. The second stage has five LRE's, with

a total thrust of 527-499 tons. The oxidant in the second and third stages is oxygen, and the fuel is hydrogen. The third stage has an LRE with a thrust of 94-80.2 tons.

Figure 40 shows a diagram of one of the first stage LRE's. The engine unit of this stage consists of five combustion chambers, four of which are arranged in a circle and capable of swiveling, while one is fixed, in the center. Each combustion chamber is made of two series of shaped steel tubes. The head of the standard engine has an atomizer to supply the fuel and the oxidant. The combustion chamber cooling system is regenerative. Fuel from the turbo-pump unit (TPU) reaches the nozzle along the inner row of tubes and returns to the head of the engine along the outer row of tubes, where it is supplied to the atomizers via radially located channels. The temperature of the gases formed during combustion of the fuel components in the combustion chamber exceeds 3000 C. /97

The nozzle jet is bell-shaped. The nozzle walls, like the combustion chamber walls, are made up from two rows of tubes. The nozzle is cooled by the fuel.

The turbo-pump unit has an oxygen pump and a fuel pump. The turbine capacity is 60,000 liter-sec. The diameter of the turbo-pump unit is 1.22 m, and its mass is 1270 kg. The TPU is switched on by means of a starter motor.

The gas generator of the TPU operates on the basic propellant with excess fuel and consumes 2% of the propellant used by the motor. Spark plugs are used to ignite the fuel in the gas generator. The ignition system for the engine chamber uses triethyl aluminum in combination with liquid oxygen.

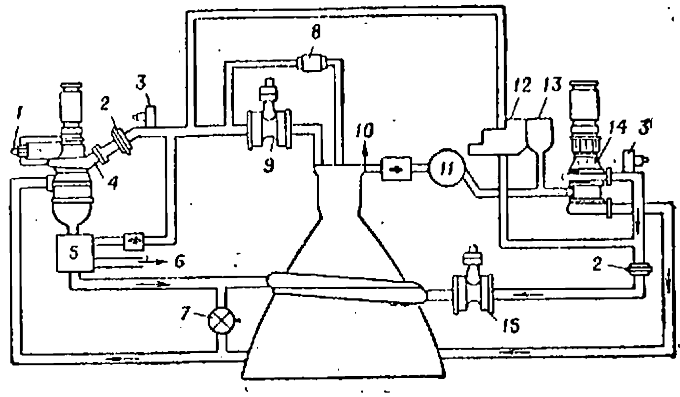

Figure 40. Schematic of the Rocketdyne liquid rocket motor of the "Saturn 5" rocket.

1- discrete individual regulator for emptying the tanks; 2- mass flow meters; 3- propellant bypass valves for cooling the supply mains; 4- liquid oxygen pump; 5- heat-exchanger; 6- oxygen for tank overpressure; 7- bypass valve; 8- oxidant supply valve in the ignition unit; 9- main oxidant valve; 10- hydrogen for tank overpressure; 11- tank of gaseous hydrogen; 12- gas generating valves; 13- gas generator; 14- liquid hydrogen pump; 15- main fuel valve.

PROPELLANTS FOR LIQUID ROCKET ENGINES

As has been mentioned, the propellant for LRE's consists of two components: oxidant and fuel.

The oxidant is the name given to the substance or mixture of substances containing predominantly oxidizing elements. The mass of oxidant is roughly two-thirds of the total propellant mass.

The fuel is the name given to the substance or mixture of substances containing the predominantly combustible elements.

Rocket fuel oxidants fall into two groups: low-boiling and high-boiling. Liquid oxygen is the low-boiling oxidant in

practical use at present. Liquid fluorine and ozone are in the development stages. Amongst the high-boiling oxidants in wide use are mixtures of nitric acid, tetranitromethane, nitrogen /98 tetroxide, and hydrogen peroxide. The possibility of using chloric acid and certain chlorine compounds is being investigated.

Liquid oxygen is obtained by successive compression of air and removal of nitrogen and other gases entering into the composition of the Earth's atmosphere. Pure oxygen does not explode, neither in the gaseous nor in the liquid state, but mixtures of it with small amounts of fuel are capable of being exploded even by very weak pulses. Therefore, the tanks, pipes, and fittings, both for liquid and gaseous oxygen, must be carefully degreased.

In order to reduce loss, liquid oxygen is stored and transported in special thermally-insulated containers. Various metals behave in different ways at liquid oxygen temperature. Copper, brass, aluminum and their alloys and steel alloy show practically no change of property, and can therefore be used to make the tanks, tubes, and fittings for liquid oxygen. Ferrous metals lose their ductility and become quite brittle. The packing materials used in liquid oxygen equipment are those which show very little change of elasticity at low temperature.

Liquid ozone has a dark blue color. Ozone is an unstable substance which tends to explosive decomposition under the action of external impulses, and this tendency increases when various impurities are present. The explosion hazard from ozone is reduced when it is diluted with oxygen, in which it dissolves very readily. The boiling temperature of ozone is -112 C, and that of oxygen is -183 C, and therefore, this solution is separated during storage: it forms two layers of which the lower has the higher concentration of ozone and has a high degree of explosion hazard.

Ozone has a high chemical activity. Many organic substances ignite spontaneously upon contact with it, while metals (all except for gold, platinum, and iridium) are rapidly oxidized.

Ozone belongs to the class of intense oxidizing substances.

Liquid fluorine has a bright yellow color and is a very active oxidizing element. Steel and copper tubes become covered on their inner surfaces by a thin but quite hard film of metal fluoride under the action of gaseous fluorine, and this inhibits further decomposition. However, the film of metal fluoride flakes off as powder at bends in the tubing and is carried away by the stream of fluorine. The least subject to the corrosive action of fluorine are nickel and certain alloys. Copper and aluminum react more slowly than the others with fluorine. Fluorine is also a highly oxidizing substance.

The fuel component of rocket propellants.

Fuels for LRE's include hydrogen, ammonia, amines, hydrazine, dimethyl hydrazine, alcohols, and kerosene. We shall examine some of these. /99

Hydrogen is one of the most promising of fuels. Gaseous hydrogen is a light gas without color or smell. At elevated temperature, it easily penetrates through glass, quartz, and certain metals. Liquid hydrogen is a colorless mobile transparent liquid.

In the pure form, hydrogen has no explosive hazard. However, mixtures of it with oxygen and air over a wide range of ratios are capable of exploding under the action of a thermal or electrical impulse. Its boiling temperature is -252 C, and its freezing temperature is -259.2 C. Since the critical temperature

of hydrogen is very low, it can be stored in the liquid form only in evacuated containers. Hydrogen is nontoxic, but inhalation of it in large amounts can lead to asphyxiation.

Hydrazine is a derivative of ammonia. It is a colorless, transparent hygroscopic liquid with the characteristic small of ammonia, and it easily decomposes under the action of various catalysts (carbon or asbestos) or when heated. When heated to 230 C, hydrazine explodes. Its flash temperature in air is 38 C. Hydrazine mixes well with water, alcohols, and amines. A mixture of hydrazine with dimethyl hydrazine is used as a fuel for rocket propellants.

Hydrazine has a corrosive property. Aluminum and its alloys, and stainless steels are resistant to it. Polythene can be used as a packing material.

Hydrazine is toxic, reacts with the mucous membrane of the eyes, and can create temporary blindness.

Fuels based on oil derivatives. Oil is the most prolific natural source of hydrocarbon fuel. Rocket motors use the kerosene fractions of oil. To improve their physical, chemical, energetic and coolant properties, kerosene fractions are subjected to a variety of processes. Fuels based on oil derivatives are noncorrosive, nontoxic and can be handled safely.

ENGINES OF THE APOLLO AND ORBITER GROUP

Some of the features of design and selection of the LRE engine unit for contemporary spacecraft can be illustrated via the example of the Apollo (Figure 41).

The engine unit for this spacecraft was subject to rigid requirements, in particular, those of lengthy storage and operation in conditions of space and weightlessness without leakage of propellant, long-term proximity of the materials and operation of the engine unit over a wide range of conditions. The design limits for the combustion chamber pressure, the composition of the mixture, and the temperature of the components were chosen to allow malfunction of certain elements and a lifetime in space in excess of the given period. During testing and shake- /100
down of the engines, investigations were made into saturation of the fuel components with different amounts of helium, which can be dissolved in them and causes low-frequency fluctuations in the chamber and the fuel lines, and these fluctuations present a hazard for the engine. The lengthy shakedown required that the chamber be equipped with a new ablative protection system. The regenerative chamber cooling system did not ensure the required removal of heat in the controlled thrust engines of the descent stage of the lunar module. A good deal of work was done to overcome difficulties associated with unstable combustion, in particular, in the first version of the engines for the service module and the flight stage of the lunar module, which did not have flame stabilization, the result being that high-frequency fluctuations occurred in the combustion chamber. Therefore, the present engines of the service module (the engine and auxiliary module) has an atomizer head with a five-element ring stabilizer, while the LRE head of the flight stage is equipped with a three-element stabilizer and a central ring. It was also difficult to provide a displacement system for supply of fuel, particularly because of corrosion arising when there were stresses in the titanium fuel tanks, and also when the materials were combined with the fuel components. There were cases where the tanks disintegrated at the design loading, owing to the action of corrosion. The most severe disintegration took place when titanium was in contact

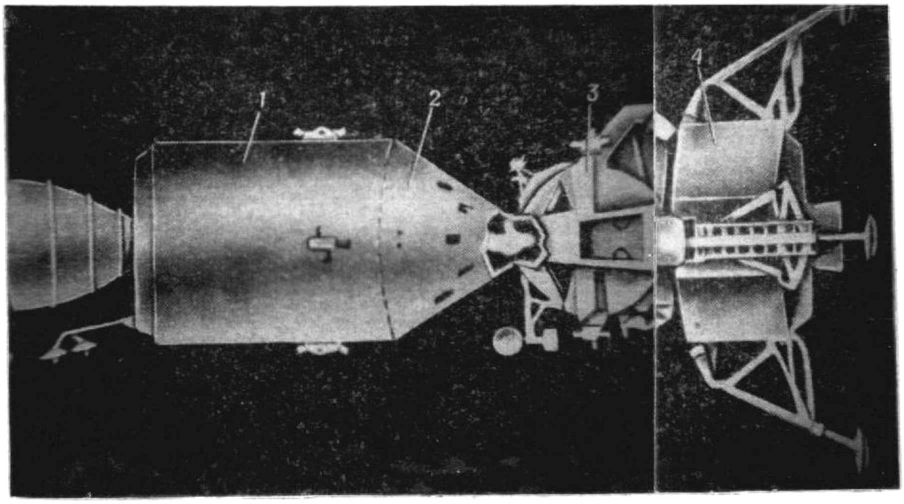

Figure 41. The Apollo spacecraft (drawing):
1- engine compartment; 2- crew compartment; 3- lunar cabin flight stage; 4- lunar cabin landing stage.

with nitrogen tetroxide (an oxidant). But if nitrous oxide was added, then no breakdown of the tanks occurred (there was no corrosion with nitrous oxide present). The simulating and flush- /101 ing liquids (methyl alcohol) also proved to be very active.

In addition, it turned out that many materials considered to be compatible have a limited useful lifetime. Leakage of fuel was observed at the flange seals, tank outlet seals, and other places. The most unfavorable event was the action of the vapor of nitrogen tetroxide in the presence of moisture, which does not exist in space, but this effect was possible on the launch pad, since the systems are filled for eight hours prior to launch of the spacecraft. All the engines of the lunar module and the main module, consisting of the auxiliary engine module (equipment module) and the crew module, operate with a single propellant. The fuel is a mixture of dimethyl hydrazine with hydrazine, and the oxidant is nitrogen tetroxide. Engine ignition results from contact between these components, which are self-igniting. The

fuel tanks are pressurized with helium, and a displacement system for supply of fuel and oxidant is used. This supply system was chosen in order to simplify the engine construction (it has no pumps, moving parts, or supplementary regulators) and to increase the operational reliability.

The thrust of the auxiliary service module engine unit is used to correct the trajectory during the flight to the Moon, for injection into orbit around the Moon, transfer to the flight trajectory to Earth, and trajectory correction during the return to Earth. This engine consists of tanks of oxidant and fuel, a helium bottle and various regulating valves and systems. The helium used to displace the fuel components is contained in two spherical tanks at an initial pressure of 220 atm. When the engine is switched off, the helium system is isolated from the fuel and oxygen tanks by means of two valves. Heat exchangers preheat the helium roughly to the temperature of the components in the tanks.

The thrust of the lunar module engine is used for the landing of the lunar module on the surface of the Moon (Figure 42). This engine has a throttled liquid LRE. The propellant components are also displaced by helium, which is contained in a supercritical state in a cryogenic tank. The fuel and oxidant are contained in parallel tanks. The tanks are equipped with sumps to avoid interruption of the supply of the components if the loads go to zero or become negative. The LRE is mounted on a swivel.

The engine of the flight stage provides thrust for the flight from the lunar surface and the rendezvous between the flight stage and the lunar module with the command and service (main) module. The helium is contained in the module in two tanks, and each propellant component is contained in a spherical tank equipped with anti-swirl devices and sumps to ensure continuity of feed at

different levels of loading. The LRE has no swivel mount, and the thrust vector is adjusted by means of the attitude control system. In the design of this unit, a good deal of attention was given to /103 ensuring speed of response of the descent stage engine to movement of the throttle, to cut=off of the thrust when the engine is cut, and to matching the dynamic characteristics of the engine and the control system. These decide the controllability of the lunar module in its maneuvers in the descent to and landing on the Moon.

At present consideration is being given to the question of what kind of motor should be used, not only for a launch vehicle, but also for a future orbiting aircraft. A liquid rocket engine is suggested, with cryogenic propellant components and a thrust of 123.5 tons. During tests of a model of this engine, combustion chambers were tested with evaporative and regenerative cooling methods. Further tests will be done with evaporatively cooled chambers, since this cooling system is simpler and ensures a longer life of the motor.

The engine has low and high pressure TPU's. The low-pressure pumps pump hydrogen which is first used as a coolant for the main nozzle, and is then expanded in the turbines. The high-pressure pumps pump the combustion products of hydrogen in the fore-chamber. All the fuel, apart from a small amount used for cooling and balancing the thrust, is burned in the fore-chamber with approximately 15% of oxygen. The combustion products, enriched with fuel, and having a medium temperature, are first expanded into the main turbines, which turn the high-pressure pumps, and then these products are burned with the remaining part of the oxidant in the main combustion chamber.

The walls of the main combustion chamber are cooled by evaporation of a small amount of hydrogen which arrives along

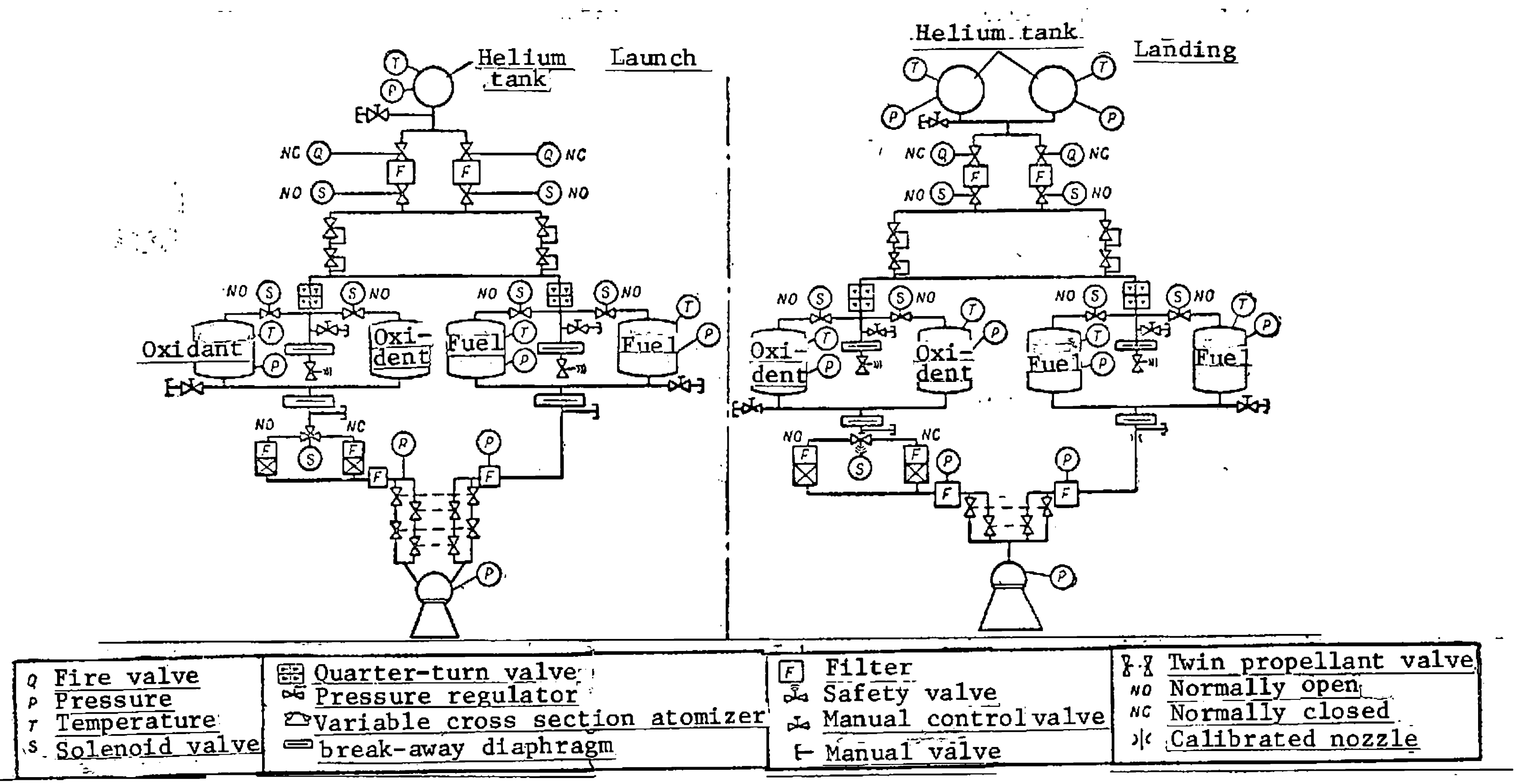

Figure 42. Schematic illustration of the engine units of the Apollo spacecraft lunar module.

the thermally stressed side of the wall (Figure 43). The main nozzle is cooled regeneratively by hydrogen passing through a staggered channel heat-exchanger. Hydrogen is supplied to a volume in front of the main atomizing head through the hydrogen outlet tube in the main nozzle, for evaporative cooling of the main combustion chamber. The remaining part of the main nozzle coolant is used as a working substance to drive the low-pressure pumps. Then the coolant is mixed upstream of the main head with the working substance of the high-pressure turbines and is supplied to the main combustion chamber.

The engines of the orbiting stage differ from those of the accelerating stage only in having a longer nozzle, required to increase the engine efficiency when operating at high altitude. The mass of the accelerating stage engine is roughly 3150 kg, and that of the orbiter stage is 3700 kg, while the specific impulse values are 442 and 465, respectively. The engine start-up is designed in accordance with the operating time, with advance supply of oxidant to the igniter and the fore-chamber, to ensure reliable ignition of the propellant. In order to achieve normal mass flow of oxidant in the liquid phase and to avoid a sudden increase in temperature in the turbine because of change in the phase state of the oxidant, the oxidant cavity in the fore-chamber head is filled before opening the fuel cut-off valve. /104

Operation of the main engine stage is controlled by four regulator valves: the fore-chamber fuel valve, the fore-chamber oxidant valve, the main oxidant valve, and the evaporative coolant system valve. To achieve safe operation of all elements, the circuits of the closed regulators are energized constantly. When the fuel is to be cut off, the fore-chamber oxidant valve is closed first, and then the fuel supply system.

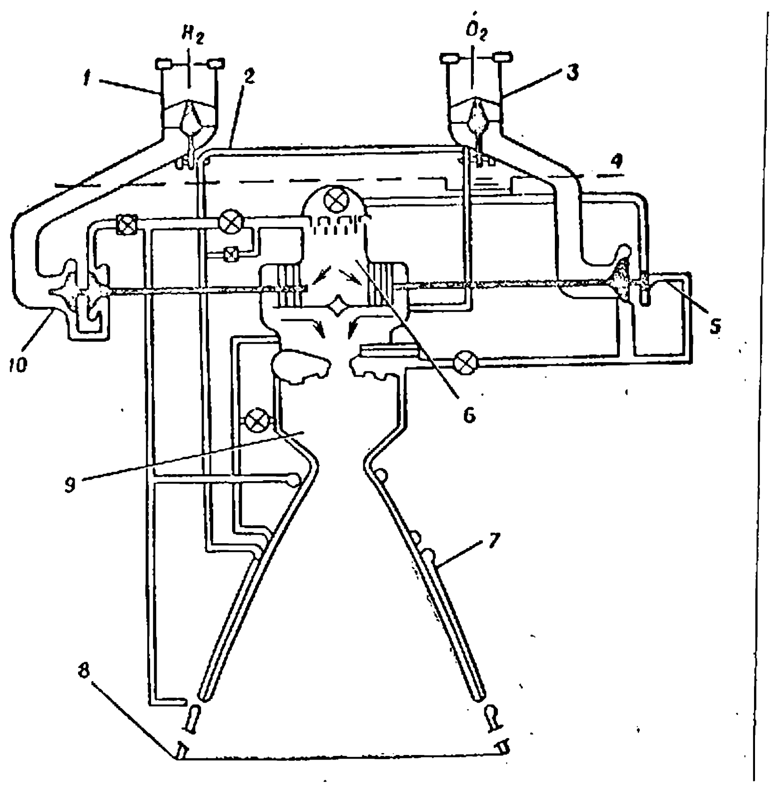

Figure 43. Arrangement of the elements.

1- low-pressure fuel turbo-pump; 2- low-pressure turbine working substance; 3- low-pressure oxidant turbo-pump; 4- Cardan suspension plane; 5- high-pressure oxidant turbo-pump; 6- fore-chamber; 7- regeneratively cooled main nozzle; 8- return-flow cooled nozzle; 9- main chamber with evaporative cooler; 10- high-pressure fuel turbo-pump.

The first start-up of the motor is presumed to occur before the aircraft structure is fully designed: the plan is that the motor will have a certain strength, to enable it to be mounted both on the accelerating stage and on the orbiter stage.

The accelerating stage engines will be operated in conditions corresponding to those at sea level. Any increase in the mass of the orbiter craft must clearly be accompanied by a corresponding increase in the thrust of the engine on the ground. It is assumed that if there is an increase in the channel area /105

in the turbine within the range of the existing class of nozzle unit blades, and if there is an increase in the nozzle throat diameter by 6.35 mm, the engine thrust will be increased up to 272.4 ton.sec.

The engines of the orbiter stage can be operated under various conditions, which depend on whether the engines are operated before stage separation or after this, in normal conditions or in an emergency situation. To obtain a satisfactory start-up in weightless conditions, the oxidant is supplied to the engine with an initial small acceleration, which ensures that the time spent by the fuel in the combustion zone is a maximum, and ensures reliable start-up.

For missions to the planets of the solar system in spacecraft, it is assumed that one will use LRE's which work with oxygen fluoride and diborane. The use of these LRE's is one way to increase the payload. With 20 start-ups, the service lifetime of an engine is ten years. It is considered that with a comparatively low pressure in the combustion chamber, and also with a rather wide range of temperature, in which the two propellant components are stored in the liquid state, storage of the components is simplified in the conditions of a lengthy space flight. In addition, because of the low pressure in the LRE, it will be possible to use a discharge supply system instead of a pump system, which considerably simplifies the construction and increases the reliability. The duration of operation of an engine unit in a mission to Mars must be 1066 days, and of one to Jupiter must be 1074 days. In this time, there will be 4 or 5 starts of the engine, which will make corrections to the flight trajectory and transfer the spacecraft into an artificial satellite orbit of the planet.

SOLID FUEL ROCKET ENGINES (SFRE)

Solid rocket engines (Figure 44) are widely used in space vehicles. The propellant used in these is a powder charge, which is located within a combustion chamber in a previously determined form. The basic distinctive feature of the solid rocket engine is its simplicity of construction, arising from the absence of such moving parts as valves, turbo-pumps, etc. After a solid engine has been started, the combustion usually proceeds until all the propellant is burned. Another peculiarity of this kind of engine is that the law for the variation of thrust is determined beforehand, and practically no control can be applied without very complex arrangements.

Solid rocket engines can be used with advantage to create a large thrust for a small time (e.g., in an emergency escape system during a launch). On the American Mercury Spacecraft, one /106 of the important functions, that of deorbiting, was performed using a solid rocket motor. The range of application of solid motors is wide: from engines with very low impulse, used for attitude control of a spacecraft in rotation, or for separating rocket stages, to engines of medium and large size, e.g., to accelerate ballistic rockets of the Minuteman type.

A solid propellant has the outward appearance of a resin or a soft plastic. The propellant contains all the substances necessary to support combustion.

In a powdered rocket mixture, the oxidant and fuel are carefully mixed, usually as a powdered combination. In some cases, the propellant is prepared by adding a powdered oxidant to a liquid fuel, the mixture then being well stirred, poured into an appropriate mold, and allowed to congeal and set.

In other cases, the solidifying mixture of fuel and powdered oxidant is crushed, pressed, and then stamped to create a powder charge of the necessary shape.

Ordinary powdered mixtures have a specific impulse of 220 to 250 sec. By adding high-calorie metallic elements, one can increase the specific impulse by 10 to 20 sec. However, as the literature abroad shows, this sometimes leads to erosion of the nozzle and even to a sharp short-duration increase in the combustion chamber pressure, which can lead to disintegration of the engine.

It is known that the thrust force of a rocket motor is the reaction arising in the engine structure due to discharge of the combustion products from the nozzle. The momentum is defined to be the product of the mass and the speed.

The burning surface area of the charge is one of the dominant parameters in the characteristics of a SFRE. By keeping the burning surface area constant, one can create the maximum allowed pressure in the combustion chamber, and therefore, ensure a maximum thrust for the whole operating time of the engine. Powdered charges can be end-type and external or internal combustion types. The charge of an end-combustion type burns only in a direction parallel to the longitudinal axis, the charge of an external burning (rod) type burns only along the external surface, and the charge of an internal combustion type (tubular) burns only on the inside surface of the channel.

An end-type combustion charge fills the entire cylindrical part of the combustion chamber, and only its back face remains exposed. With this form of construction of the charge, the volume of the combustion chamber is filled 100%, and the combustion surface area is always constant. However, a defect of the end-type

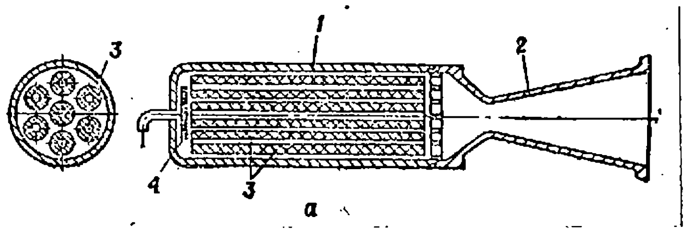

Figure 44. Schematic of a solid-fuel rocket motor.
1- combustion chamber; 2- nozzle discharge nozzle; 3- solid fuel; 4- igniter.

combustion charge is that it generates comparatively small thrust. If the ratio of the charge length to its diameter is greater than unity, the thrust is determined by the cross-sectional area of the charge. The thrust can be increased by increasing the chamber diameter. However, the aerodynamic drag then increases appreciably. In addition, when using end-combustion charges, one must introduce thermal insulation or increase the thickness of the chamber walls, in order to protect the chamber walls from being heated by the hot combustion products.

A rod-type charge is an ordinary bar of propellant, which burns over its external surface. Because of the large combustion surface area, the rod charge generates a larger thrust than the end-type combustion charge. A defect of this form of SFRE is that the chamber walls are not protected from the action of high-temperature gases. Because of this, the rod charge has a tendency to sag or settle before the start of burning, to adopt a wavy curve during burning, and to break up immediately before the end of combustion.

Tubular charge To avoid deformation of the charge and overheating of the combustion chamber walls, tubular charges are used which turn over the inner surface. For this kind of charge, the duration of burning and the degree of volumetric saturation of the combustion chamber are the same as for the rod charge. With a tubular charge, special thermal protection of the combustion chamber walls is not required and no charge deformation deforma- occurs. However, in this case the combustion chamber walls must be designed for a maximum pressure, corresponding to the maximum combustion surface area.

Combination tubular-rod charges and charges with a star-shaped channel section are used.

Structurally the combustion chamber usually takes the form of a thin-walled cylindrical vessel, operating at high pressure. The body of the combustion chamber is made principally of steel alloy, and in recent years titanium and fiberglass have been used, but, as the literature abroad indicates, only steel alloys have been used in spaceships, for reasons of greater reliability.

In long-time engines, the forward and rearward parts of the combustion chamber are covered with asbestos or resin, which have good thermal insulation properties. If coarse charges are used, or if the motor is subjected to the effect of considerable temperature gradients, which produce a large curvature of the charge, the insulation is made in two parts, joined by a special sheath. This sheath is fastened to the combustion chamber wall, allowing a certain gap which allows the charge to change its dimensions as its temperature changes, without causing stress in the structure. /108

The engine nozzle is usually made of steel with a graphite coating at the throat. The nozzle structure uses sealing rings to prevent leakage of gas in the joint between the combustion flange and the nozzle. The expanding part of the nozzle is made of an ablative material. The ignition must generate a temperature at the charge surface in excess of that required to ignite the charge, and raise the combustion chamber pressure above the minimum value required for stable combustion. To ignite the propellant surface requires a certain time during which heat must be supplied. The heat supply should be continued until the rate of heat loss from the charge surface due to conduction is not less than the rate of supply of heat due to exothermic reactions. If this condition is not fulfilled, the propellant will not ignite. The smaller the pressure in the combustion chamber, the longer it will take to ignite. The ignitor must provide a smooth and rapid pressure variation up to the level for stable combustion. Ignition which is too fast can cause an initial pressure rise, sharp shocks, and splitting of the charge. Conversely, if the pressure is insufficient, surges in the thrust, with a flare-up and collapse can take place, and also the charge may simply not ignite.

Two types of igniter are used: pyrotechnic and gas generating. Pyrotechnic igniters are composed of granules of a mixture of metals and oxidants. The igniter is ignited by a fuse which then ignites the auxiliary charge, which is usually made from a finely-ground pyrotechnic compound.

The gas generating igniter is essentially a small rocket engine and is used to start up the main engine. The gas generator has a small pyrotechnic igniter to ignite the propellant charge, has a good ignition capability, and generates a large amount of gas. The small engine operates at very high pressure, and the gases discharging from its nozzle ignite the main rocket engine charge.

As an example of one of the high power SFRE's, we can take the motor for the Titan rocket (Figure 45). Its thrust is 450-553 ton. sec, the burn time of the propellant charge is 100-120 sec, its length is 25 m, and its mass is 250 tons. The propellant charge is cylindrical with a tubular cut-out, it burns from the inside at each section and from the ends, and the thickness of the propellant layer is 900 mm. The engine case is steel, its inside surface has a rubber-like insulation, enclosed with a sealing material.

The length of the nozzle engine is 3 m and the diameter of the nozzle exit is 2.7 m. The igniter for the main charge in this engine is a small solid-fuel igniter. The thrust vestor is controlled by injecting liquid nitrogen tetroxide under pressure into the nozzle. Two apertures in the forward cover of the engine (they are covered when the engine is cut off) are used to cut off the thrust. The Titan 3C rocket uses two such high power lateral motors.

The powerful first stage space rockets in the USA are designed to use a solid-fuel rocket motor with a thrust of 1360 tons, a burn time of 60 sec, and a fuel mass of 318 tons. The diameter of the engine is 4 m, and its length is 24.4 m. The motor casing is made up of three sections. The central section has a mass of 150 tons, and a length of 6.7 m. This motor is equipped with a thrust vector control system using injection into the nozzle of liquid or gas contained in 12 tanks, which are located around the main nozzle. There are 24 pumps for injecting the liquid or gas.

The foreign literature mentions that plastics and fiberglass are widely used at present in the fabrication of solid motor casings and nozzles. The nozzle must withstand an exhaust gas temperature of up to 3595 C. The Thiokol motor (Figure 46),

designed to be the third stage of the Minuteman rocket, has a casing made of fiberglass (the casing is fabricated by a winding method). The casing of the Aerojet solid motor, mounted in the first and second stages of the Polaris rocket, is also made of fiberglass.

It should be mentioned that the American spacecraft Mercury and Gemini used solid rockets for the Earth return, while their attitude control systems used liquid motors. The Apollo spacecraft has had a liquid basic motor, while solid motors were used only in the emergency escape system to separate the command module if an emergency arose during the launch. The departure from a solid motor here is due to its low efficiency, to the fact that its characteristics are not flexible enough, and to diffi- / 110
culties in making up a power unit of this kind. In particular, solid motors, as a rule, cannot be launched repeatedly, and their possibilities with regard to thrust control are limited.

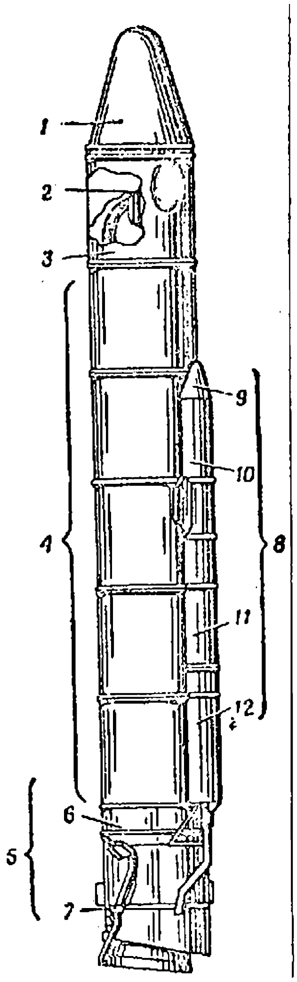

Figure 45. Schematic of the Titan solid-fuel rocket.

1- nose section; 2- ignition system unit; 3- forward section of motor; 4- main motor sections; 5- rear part of engine; 6- engine rear skirt unit; 7- ring collector for the vector thrust control system, located in the nozzle; 8- tank unit for vector thrust control system; 9- nose section; 10- tank with gaseous nitrogen under high pressure; 11,12- forward and rear nitrogen tetroxide tanks.

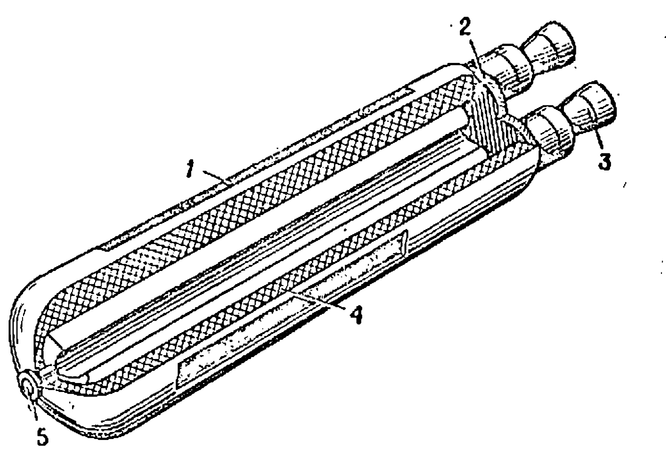

Figure 46. Schematic of the Thiokol solid-fuel engine, with a thrust of 77-91 tons, installed in a Minuteman rocket.

1- casing; 2- covering and insulation; 3- nozzle; 4- fuel charge; 5- igniter.

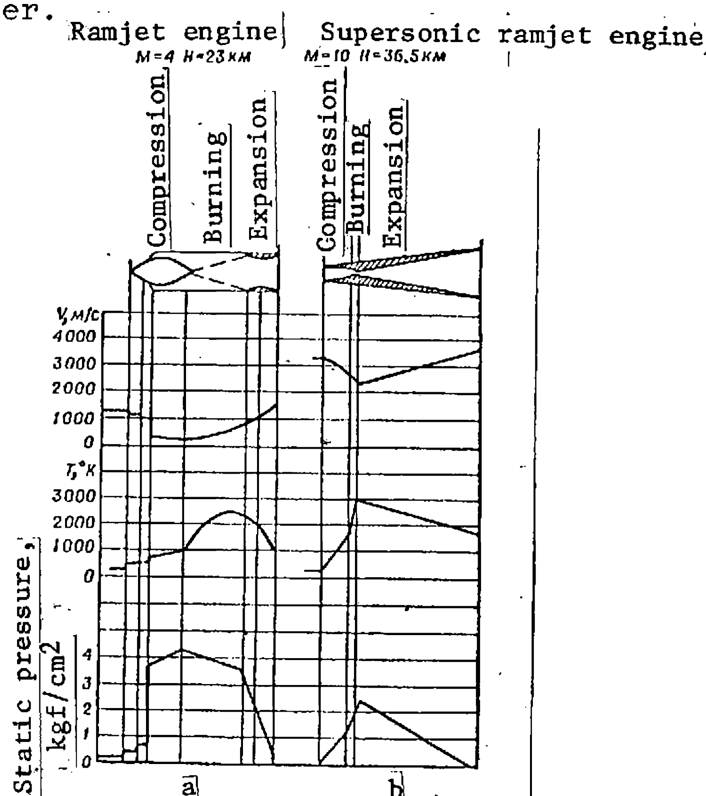

Figure 47. Variation in parameters during operation of an air-breathing rocket engine.

a- with subsonic combustion; b- with supersonic combustion.

AIR-BREATHING ENGINES

In contrast with liquid rocket motors, air-breathing motors require only fuel for operation in a vehicle, and use oxygen from the surrounding atmosphere as the oxidant. It is, therefore, possible to use them in air-space vehicles in the atmosphere.

Ramjet engines are the simplest of air-breathing engines. For an aircraft in flight with a ramjet engine, the incident air is decelerated in front of the engine entrance and in the entrance diffuser, thereby increasing its pressure (Figure 47), necessary for the operating process in the engine. The degree of pressure increase in this engine and the operation of the engine elements depends appreciably on the flight speed and the design of the inlet section.

When a supersonic air flow is decelerated in a normal shock there are large pressure losses, which appreciably reduce the pressure at the end of the compression process. Therefore, for supersonic ramjet engines which are proposed for air-space vehicles, the supersonic stream is decelerated in a special device, a supersonic air intake, in order to reduce the pressure loss at the combustion chamber entrance. In such air intakes the supersonic flow is compressed in a system of conical shocks, /111
finishing with a weak normal shock. Then several weak shocks replace the single powerful shock, and the pressure loss of the air flow is appreciably reduced. The shock system is generated when the flow passes over a special tapered body (a cone), extended upstream into the incident flow.

After compression in the intake, the air reaches the combustion chamber. There the air is heated by continuous combustion of fuel supplied through the pumps. The combustion

products, heated in the combustion chamber to a high temperature, come into the reaction nozzle, where they expand. Then their velocity is increased. The result is that the exhaust velocity is greater than the flight speed, and a thrust is created which propels the vehicle forward. However, a ramjet engine is not capable of autonomous flight (it has no thrust at low speed), and therefore, it is used in combination with other engines, or else a vehicle with a ramjet engine must be accelerated to the required flight speed.

For ramjet engines designed for flight speed up to M = 3 to 4, the flow is decelerated in the air intake to subsonic speed, i.e., the heat is supplied to a subsonic flow. As the flight speed increases to M = 5 to 8, it is desirable to supply heat to the supersonic stream, without decelerating the flow to subsonic speed. For flight speed of M = 6-7, ordinary ramjet engines are used with the air flow decelerating in the intake from supersonic to low subsonic speed (60-150 m/sec) at the entrance to the combustion chamber, in order to ensure stable burning and to avoid separation of the flame, a contingency which brings in many problems. These problems arise, in particular, at very high temperatures and pressures in the combustion chamber, at flight speeds greater than that corresponding to M = 6. The static temperature and pressure at the combustion chamber entrance are increased because of compression of the air in the engine intake to such an extent that the thrust begins to fall sharply. For M > 10, the thrust becomes negative.

Investigations of the operation of ramjet engines, conducted by engineers abroad, have shown that it is unnecessary to decelerate the flow entering the combustion chamber to subsonic speed. It was established that combustion can occur with an air flow speed in the body of the engine of 1500-3000 m/sec and that useful thrust can be obtained (at least in theory) at a

flight speed close to that corresponding to M = 25.

Depending on the mission profile and the fuel used, air ramjet engines with supersonic combustion are designed so that the air flow is decelerated sufficiently to increase the static pressure and temperature to values required for self-ignition of the fuel injected into the stream.

An engine of this kind is called a ramjet engine with supersonic combustion or a ramjet engine with supersonic combustion speed.

Ramjet engines with supersonic combustion are devices consisting of an air intake, a combustion chamber, and a nozzle. They differ from an ordinary ramjet engine in that for hypersonic flight speed the flow speed in the intake, although it is reduced, remains supersonic, so that the combustion occurs in a supersonic flow (Figure 47,b). At low hypersonic flight speeds, this kind of engine has low efficiency, but for a wide range of supersonic flight speed, its efficiency is quite large. In addition, for a considerable part of this range, one can use an engine with irregular geometry and a slight reduction in efficiency, and this considerably simplifies the engine construction. It is sometimes stated in the foreign literature that, in essence, a ramjet engine with supersonic combustion nozzle does not make sense, although the combustion chamber opens up immediately into a nozzle, forming a single unit. The absence of a nozzle throat /113
means that there is no transition of speed from subsonic to supersonic in this case, as is usual.

In addition, the efficiency of a ramjet engine with supersonic combustion becomes equal to that of an ordinary ramjet engine at M > 7, although if need be, a ramjet engine with supersonic combustion can be used as low as M = 4. This qualitative transition is due to the fact that in ordinary ramjet engines, at

large values of M the losses in the air intake increase to an acceptable degree while in a ramjet engine with supersonic combustion at low values of M, the losses in the combustion chamber increase to an unacceptable level (in comparison with the losses in ordinary subsonic combustion).

The foreign literature mentions a number of important problems which must be solved when one constructs a ramjet engine with supersonic combustion. First one must increase the speed of the air flow in the intake; this is usually decided by the ratio of the flight speed to the speed of the air in the intake. When this ratio decreases, the specific impulse of the engine increases. This is due to the fact that the losses in the intake decrease and the temperature of the gas heated by compression at the entrance to the combustion chamber is reduced.

If the air intake is designed for the best speed ratio, then the static temperature at the combustion chamber entrance is approximately five times the temperature of the surrounding air (for $M > 10$), and exceeds 830 C at all heights, which means that the fuel injected into the air will self-ignite.

The time required for the combustion reation, i.e., burning of the fuel in air, has been the subject of many investigations.

Mixing of fuel with air, as is well known, involves molecular diffusion, turbulent vortices, and macro-mixing resulting from vortex effects. Even diffusion of such fuels as hydrogen proceeds rather slowly and does not provide appreciable mixing of the hydrogen and air during the time they are in the combustion chamber of a ramjet engine with supersonic combustion. Studies of turbulent mixing have shown that the length required for satisfactory premixing of the fuel and hydrogen, is roughly equal to 100 jet diameters, and that macro-mixing becomes more effective

when centrifugal atomizers are used.

The problem of achieving subsonic and supersonic combustion in a single channel has also been solved. At low values of M the air temperature will be too low for self-ignition of the fuel. Therefore, it is possible for subsonic combustion to require the usual system for ignition and stabilization of the flame, and for this not to be required in supersonic combustion without stabilization at large enough flight speed, where the air temperature is increased. The foreign literature mentions that the main difficulties in achieving burning will be associated with the regime /114 of transition from subsonic combustion to supersonic, since as the flight speed of a vehicle with a ramjet engine increases, the air flowing through the engine channel can remove the flame until the flight speed and the air temperature become so large that good mixing of the fuel is ensured.

Engineers abroad are studying various fuels for ramjet engines with supersonic combustion. The main interest is in liquid hydrogen. Contemporary heat-resisting metal alloys withstand steady temperatures of about 1000-1300 C. Temperatures like this are found in flight at speeds corresponding to M = 8-12, at a height of 30-36 km. Therefore, the opinion is expressed that before creating new materials for vehicles with ramjet engines, it will be necessary to cool the leading edges of the wing, certain parts of the air intake, the combustion chamber, and the exit nozzle.

A defect of hydrogen is its low density, and therefore possibilities have been studied for using other fuel for engines of this kind. Theoretically ordinary hydrocarbon fuels are suitable for ramjet engines, but in practice the possibility of using these fuels depends primarily on their self-ignition temperature.

One of the main advantages of any ramjet engine is that it uses atmospheric oxygen as an oxidant and it therefore achieves higher values of specific impulse than rocket engines do. The best contemporary rocket engines use 5-6 kg of oxidant for 1 kg of fuel, with a specific impulse of 400 sec. Design data for ramjet engines with supersonic combustion, operating with hydrogen, show that a specific impulse may be achieved up to 1000 sec at M = 20, and up to 4000 for M = 4. Ramjet engine operating with kerosene can develop a specific impulse of up to 400 sec at M = 15, and up to 1200 sec at M = 4. Thus, the ramjet engine with supersonic combustion consumes 1/10th of the mass of fuel at M = 4, 1/5th at M = 10, and 1/12th at M = 20, in comparison with the best contemporary liquid engines operating with liquid hydrogen and liquid oxygen.

However, as regards thrust, ramjet engines with supersonic combustion fall below a rocket motor. The thrust of a ramjet engine depends on the Mach number and the flight height. A ramjet engine cannot operate in vacuum, like a rocket motor. A ramjet engine can develop large specific thrust only during flight in a dense atmosphere. As the flight altitude increases, to maintain its thrust, a ramjet engine must either increase the flight speed, or expand the inlet section of the air intake, in order to increase the mass flow of air through the engine. At an altitude of 48 km, there is still sufficient air to maintain supersonic combustion and create the necessary thrust. /115

At large flight altitude the resistance and aerodynamic heating of a vehicle diminish, while the flight speed can be increased. For this reason, the most favorable speed for a vehicle with a ramjet engine is that corresponding to M = 10 at a height of 30 km or M = 20 at a height of 45-48 km.

Thus, many problems must be solved before a commercial ramjet engine with supersonic combustion can be constructed. The USA intends to use a greatly modified X-15 aircraft as a flying laboratory to investigate hypersonic ramjet engines. The USA is also studying the possible development of a hypersonic experimental aircraft, designed for M = 12, with a ramjet engine with supersonic combustion to replace the X-15. This experimental aircraft would be launched from a B-52 aircraft or from the supersonic B-70 aircraft. The tests will be conducted in the range M = 2-8 and even up to M = 12-15. During the flights they will investigate the operation of an engine with supersonic combustion at M > 4. For this purpose, they propose to develop a non-standard air intake, optimized to operate in supersonic combustion, but having quite good characteristics for subsonic combustion, in order to reduce the number of moving parts in the engine. The aircraft must have the help of a solid motor to fly. The flow of air of the solid motor will be controlled by flaps. At speeds corresponding to M = 5, the ramjet engine will be switched on, and will be used to accelerate the aircraft to a speed corresponding to M = 12.

In a similar fashion to what is done in one of the experimental French aircraft, an ordinary ramjet engine with subsonic combustion will be combined with a solid motor in a turbo-ramjet power unit, and a ramjet with supersonic combustion can be combined with a rocket motor to create a power unit capable of operating at speeds from low subsonic to orbital speeds. Such a unit, as is mentioned in the literature abroad, is not only theoretically possible, but is based on investigations conducted by means of computers.

Turbojet engines can be used in a considerable range of aircraft speed. The engine consists of an intake unit, a compressor,

a combustion chamber, a gas turbine, and an exit nozzle (Figure 48). Atmospheric air flows into the intake unit, where it experiences a slight compression due to the dynamic head. Then the air flow is directed into the compressor, which increases the air pressure. From the compressor, the compressed air is supplied to the combustion chamber. There finely-divided fuel is injected through jets. The fuel-air mixture, in burning, forms the operating substance, the hot gases.

Leaving the combustion chamber, the gases cause the turbine to rotate and then exit from the jet nozzle with high speed. Thus, the thrust is formed. The energy of the combustion products, acquired during compression of the air in the compressor and the heat subsequently supplied during the combustion process, is partially used in the turbine and goes to drive the compressor and all the units serving the engine. Another part of the energy, unused in the turbine, is transformed into kinetic energy in the /116
jet nozzle during the expansion of the gases. The result is that the discharge of gas from the engine takes place at high speed and creates a jet thrust. The turbojet engine is capable of creating a very great thrust power, which increases in proportion to the flight speed.

All contemporary aircraft use turbojet engines as their power unit. A motor of this kind could evidently be used to create an air-space vehicle capable of independent flight in the atmosphere.

NUCLEAR ROCKET ENGINES

The first experimental reactors confirmed that it was possible in principle to create reactors suitable for nuclear rocket engines. These reactors, with graphite moderators and uranium fuel, heated hydrogen to high temperature and created a jet thrust when the gas discharged from a nozzle.

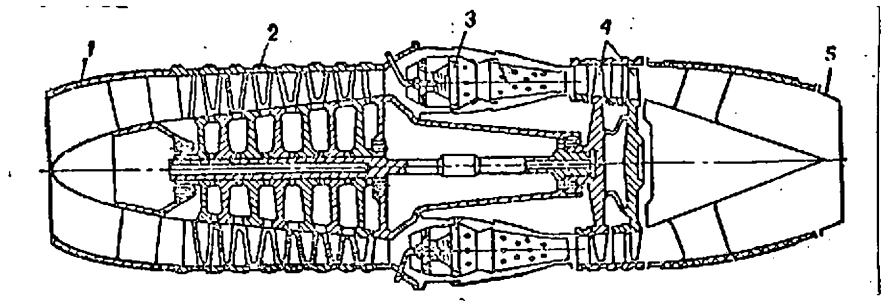

Figure 48. Schematic of an engine with an axial compressor.

1- intake; 2- axial compressor; 3- combustion chamber; 4- two-stage turbine; 5- exit nozzle.

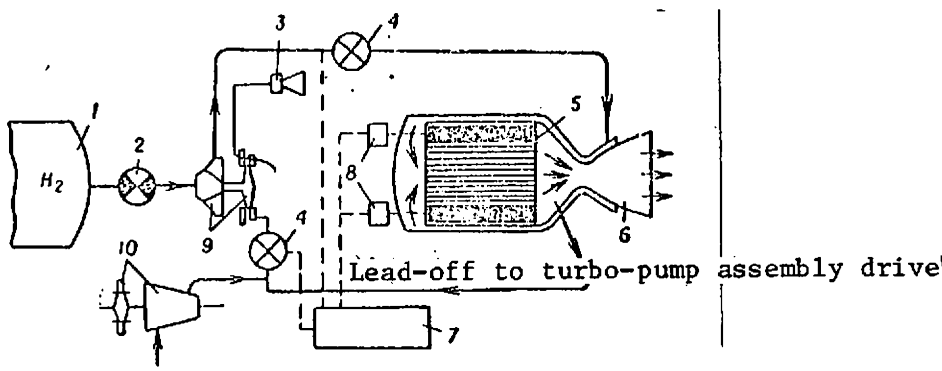

Figure 49. Schematic of a solid-fuel nuclear rocket engine.

1- tank with working substance, hydrogen; 2- shut-off valve; 3- correction engine nozzle; 4- control valves; 5- reactor; 6- nozzle; 7- control equipment; 8- control rod actuators; 9- turbo-pump assembly; 10- starting unit.

The main stimulus for the development of a nuclear engine is the hope of using the huge amount of energy released in nuclear fission for displacement of a spacecraft.

Nuclear rocket engines possess much greater energy efficiency than the thermochemical rocket motors presently used. This allows the range of applications of spacecraft of different kinds to be greatly expanded. A vehicle with a nuclear engine, creating larger thrust (in comparison with other vehicles) for the same fuel rate, could carry a greater payload to the same distance or a similar payload to greater distances.

A solid-fuel nuclear engine (Figure 49) consists of a reactor 5, a turbo-pump unit 9, a nozzle 6, a tank 1 with the working substance, the control equipment and other units. The reactor core is composed of solid heat-generating elements. The motor is equipped with a control system and an emergency reactor shield, a system for supplying the working substance, and certain auxiliary units. /117

The working substance (in this case hydrogen) is supplied from the tank 1 by the turbo-pump unit 9 to the nozzle cooling system and the reactor case, and then to the reactor 5. The working substance is heated in the reactor core, and expanding in the nozzle, is discharged from the nozzle into the atmosphere. In this way, the jet thrust is created.

The main difficulty in the development of a reactor in a nuclear engine, in the opinion of one of the American specialists in this field, is to create an airborne form of compact reactor (not much bigger than a writing desk), generating power equal to that of an electric power station, for a number of minutes following the launch. Intensive work in this field in recent years has resulted in the fabrication of reactors of this kind, as mentioned in the press. In 1969 successful tests were carried

out on the most recent experimental model of the Nerva reactor at a nuclear engine development station (Figure 50).

The tests were carried out for more than eight days, during which time the engine was started 28 times. At full thrust (about 25 tons) and full power of 1100 MW the engine operated for 3.5 min. The total time of operation of the engine at different power levels was 3 hours, 48 minutes. /118

Subsequently attention was focused on the development and tests of a flying version of this engine. Its application to a spacecraft is projected for 1977. It is known that nuclear rocket motors can create an exhaust gas speed of twice that of thermochemical engines, and can therefore provide twice the impulse. This is the main advantage, since the greater specific impulse makes it possible to conduct a space mission with less fuel, or to accelerate the same payload to a larger speed, to deliver it to a greater distance, to perform more complex maneuvers for the same amount of fuel, or to combine these possibilities in the best way.

The theoretical explanation for the fact that the discharge speed, and therefore, the specific impulse of a rocket motor of a nuclear engine is greater than the same parameters for a thermochemical motor is the following: v (or j_{sp}) is proportional to $\sqrt{T/M}$, where v is the discharge speed; T is the discharge gas temperature; and M is the average molecular weight of the gas.

It is considered at present that thermochemical engines operate already at the limiting temperatures which structural materials can sustain. Therefore, further increase in temperature can lead only to an insignificant increase in the discharge speed.

In particular, liquid rocket engines with oxygen and hydrogen, operating at a temperature of 2790 C, produce a specific impulse of 456 sec. Even a fluorine-hydrogen liquid rocket engine, operating at a temperature of 3750 C, would increase the specific impulse only by 20 sec. On the other hand, the value of M for a nuclear rocket engine is considerably less than for a liquid rocket engine, so that $\dot{v}$ and j_{sp} can be increased a good deal without increasing the temperature of the discharge gas. Also, the nuclear rocket motor heats the fuel, which is hydrogen with M = 2. The discharge from a thermochemical rocket motor nozzle is combustion products, a gas mixture with a considerably greater molecular weight than hydrogen. For proposed hydrogen-oxygen thermochemical engines, the molecular weight of the H_2O mixture at the nozzle exit is 18. Thus, in theory, at the same temperature, the discharge speed and the specific impulse for a nuclear rocket engine is larger by a factor of three than for thermochemical engines. A hydrogen-oxygen liquid rocket engine operating in vacuum achieves a specific impulse of up to 458 sec, while a nuclear rocket engine at lower temperature can produce a specific impulse greater than 900 sec. /119

However, the discharge velocity of these combustion products cannot be increased by a factor of two only at the expense of the payload which they represent; it also requires power to be increased by a factor of two to accelerate the discharge of the products. Nevertheless, this does not constitute a problem because a small quantity of nuclear fuel contains a great enough energy. The other advantage of a nuclear rocket is its capability to transform the huge potential energy of a nuclear fuel into a high discharge velocity.

A further advantage is the storability of nuclear energy. The solid fuel (graphite-uranium heating elements) is physically and chemically stable, and therefore, a spacecraft with a nuclear rocket can be injected by means of a launch vehicle into a geocentric orbit, and then can be supplied with this fuel, which will be brought by another launch vehicle (the tanker). In addition, a spacecraft with a nuclear rocket can be used repeatedly, if it can be supplied with fuel by means of orbiting tanks. One can assume that the resupply of fuel to a spacecraft with a nuclear rocket can be performed much more simply than for a spacecraft with a thermochemical engine, since the propellant for the latter consists of two components (fuel and oxidant).

Figure 51 shows the basic elements of a nuclear engine, which has undergone ground testing. The working substance (liquid hydrogen) is supplied by means of a turbo-pump from a fuel tank at the operating pressure. The hydrogen previously passes through a tube in the nozzle wall, which is thereby cooled, while the hydrogen is heated. Then the hydrogen passes through the reflector and the radiation shielding system and reaches the reactor core where it is heated to the operating temperature. Thrust results from the expansion of hydrogen in the nozzle and its discharge. A small amount of hydrogen is supplied to the turbo-pump unit.

The reactor core is composed of graphite with pores filled with uranium, and has a multitude of valves through which the hydrogen passes. The reflector surrounding the core prevents leakage of too many neutrons from the core, and thus assists in maintaining control of the essential nuclear fission reaction. The reflector contains control elements of cylindrical shape, one half being material which reflects neutrons, and the other half material which absorbs neutrons. The elements are rotated synchronously in such a way that the required neutron flux in the /120

core is maintained: the angular position of the elements determines the fission rate, and this means also the rate of heat release. The presently planned flight sample engine "Nerva" conforms almost wholly with the above description. The difference is that in the flight specimen, almost all the previously heated hydrogen passing through the reflector of the radiation shield system is first supplied to the turbine inlet before going to the core. Since hydrogen is not supplied to the turbine from the core, when this cycle is used, the same specific impulse can be achieved for a smaller core temperature.

The mean temperature of the core of the Nerva engine, as proposed, will be about 2090 C, its thrust is 34 tons, its power is more than 1500 MW, and its specific impulse is 825 sec. During development of Nerva, difficulties arose, associated with the strength of the reactor structure in the complex conditions of operation (high temperature and large power). Another important question to be resolved in constructing the engine is the need to accomplish repeated firings of the engine. Here the requirement is that the reactor withstand the repeated heating of too high temperatures. Repeated firings are required. In particular, for the Earth-Moon-Earth interorbital flights, the plan is that the motor can be switched on in the following situations: to inject the spacecraft into a trajectory to the Moon, for a possible midcourse trajectory correction, to decelerate the spacecraft for injection into a selenocentric orbit, to inject the spacecraft into a flight trajectory to Earth at a possible midcourse correction of the return trajectory, and in addition, to decelerate the spacecraft upon transfer to a geocentric orbit. For each engine start /121 the power can vary from zero to the maximum, and the operating time depends on the payload.

The next problem stemmed from possible prediction of the conditions for operation, control of the engine system, and

reliability of operation. In particular, the engine must go to full power in about 1 min, while certain industrial reactors go to maximum power in several hours.

Tests have shown that the energy capability of the Nerva system is large, including a capability of long-term use of the active elements. The high specific impulse of a nuclear engine and the great quantity of available energy stored in the nuclear fuel increase the maneuver capability of a spacecraft and expand the range of payloads, in comparison with existing and even future thermochemical rocket engines.

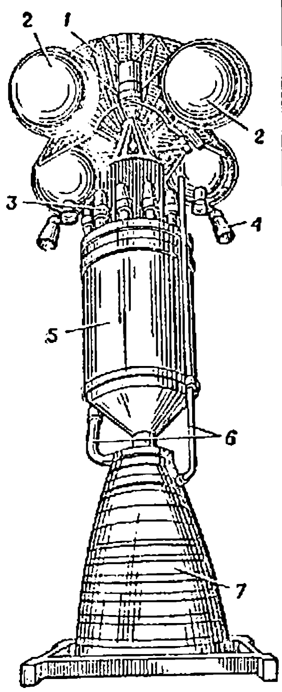

Figure 50. The Nerva experimental rocket engine.

1- frame sensitive to the thrust force; 2- spherical tanks of compressed gas; 3- pneumatic mechanism controlling the position of the reactor rods; 4- nozzle for yaw control; 5- nuclear reactor; 6- pipe supplying gas to turbo-pump assembly; 7- staggered nozzle unit.

A particularly obvious advantage of a nuclear engine appears in performing orbital transport operations in the vicinity of Earth and regular journeys between Earth and the Moon. In particular, regular journeys between the Earth and the Moon could be arranged as follows.

A stage with a nuclear engine is injected by means of a launch vehicle into a parking orbit at altitude 550 km. The interorbital transport craft, consisting of fuel tanks, service and command modules and a nuclear engine, is assembled in one parking

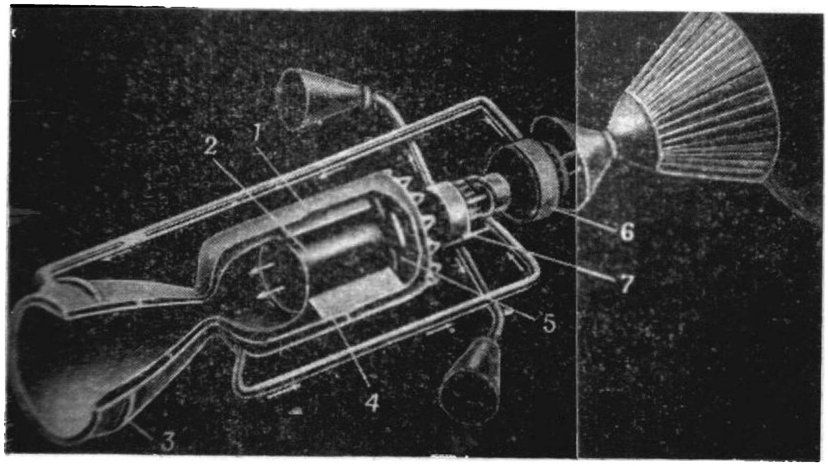

Figure 51. Simplified schematic of the Nerva experimental nuclear rocket engine.

1- thrust-bearing skin, high-pressure shell; 2- reactor core; 3- nozzle; 4- reflector; 5- shield; 6- pump; 7- turbine.

orbit and is then fueled. The transport vehicle with the nuclear engine is injected into a trajectory to the Moon and after transfer and deceleration performs a rendezvous with a stage in selenocentric orbit. After transferring its payload to the circumlunar stage, the transport leaves on the return trip. In the vicinity of Earth, the nuclear engine is again switched on and the transport transfers to a parking orbit. There it can again be refueled, and thereafter sets out on the next trip to the Moon with a new payload.

The refueling and replacement of the payload can be performed with the help of an orbiting station in geocentric orbit, which in turn is provided with all necessities by means of a multiple-use transport vehicle of the Earth-orbit-Earth class. The payload of a nuclear transport vehicle might include crew, passengers, life-support system, and also scientific equipment for investigation of the Moon.

Future applications of nuclear engines for investigation of the solar system are also being studied. Calculations have been published in the foreign literature which determine the yield of one kilogram of payload using engines of different kinds. The specific impulse for a liquid rocket engine was /122 taken as 465 sec, for a nuclear engine it was 825 sec. Omitting the complex calculations, we give the final data: if a multiple-use transport system of the "Earth-orbit-Earth" class is considered and it is used as a launch platform for a single-phase thermochemical stage of mass 182 tons, then about 18 tons of payload can be injected into the most energetically unfavorable polar geocentric orbit, having expended roughly 620 dollars per kilogram. If one plans to return the vehicle to Earth, the 18 tons must include the mass of the launched vehicle, carrying the payload of mass about 5.5 tons. The cost of this load is about 2000 dollars per kilogram beforehand.

But if the multiple-use accelerating stage is replaced by a standard launch vehicle, which would inject a multiple-use nuclear engine with a mass of 18 tons, capable of atmospheric reentry and landing, this rocket system could accomplish a similar task and return to Earth 5.5 tons of payload at a cost of about 880 dollars per kilogram.

Therefore, the foreign specialists conclude that:

1. For transportation of non-return loads (such as fuel to replenish engines) either from orbiting launch systems, or directly from the Earth's surface, it is clearly preferable to use a single-use stage with a thermochemical engine unit.

2. To transport returnable payloads to given orbits, starting from a parking orbit, it is preferable to use a ramjet vehicle with a nuclear engine: these have a clear economic advantage over

the single-use thermochemical system. An economically more efficient solution for the above operation would be to create a multiple-use transport craft with a nuclear engine, designed for direct atmospheric entry and landing.

3. For returnable payloads launched from the Earth's surface, it is desirable to make partial use of multiple-use vehicles with thermochemical engines. As in the first case, it will be most economical to use multiple-use upper stages with nuclear engines, returned to the Earth.

ELECTRICAL ROCKET ENGINES

In recent years, foreign scientists and engineers have explored the use of so-called electrical engines in space applications. It is reported in the foreign literature that these engines will be very widely used in space vehicles.

The term "electrical engine" has been used for a long period to denote only engines in which electrically charged particles or an electrically conducting working substance is accelerated by electric or magnetic forces. The idea is that the kinetic energy of the ejected particle depends on the electrostatic forces (in contrast with the thermodynamic forces, which act in rocket motors operating with chemical fuel).

In recent years, investigations have concentrated on other processes in which the electrical energy is transformed into kinetic energy of the discharging stream. It has been established by experiments with high-intensity electrical arc discharges that under favorable conditions, the heated working gas discharges from the arc chamber at supersonic speed.

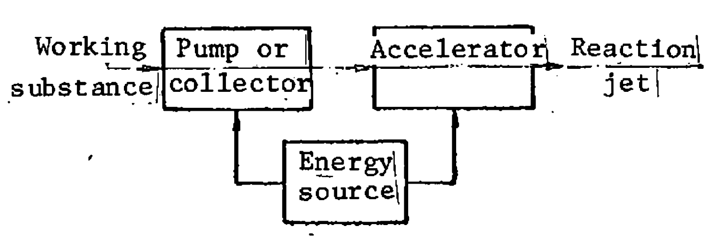

Figure 52. Schematic of electric jet engine.

In chemical-fuel rocket motors, the thermal energy is obtained from the chemical energy of the bond, released when the fuel and the oxidant react. Here the engineer has a limited choice of fuel components with high reaction energy (e.g., kerosene-oxygen of hydrogen-oxygen).

In the electrical engine (Figure 52) there must be a separate source of energy and a corresponding conversion unit to communicate the required kinetic energy to particles of the working substance. The energy source and the accelerator (converter) must be component parts of the engine producing the thrust.

The discharge speed of the gases in a rocket motor using chemical fuel is limited by the heat release capabilities of the reaction between the fuel and the oxidant. Fuels which possess the greatest stored energy (e.g., hydrogen-oxygen or hydrogen-fluorine), provide gas discharge speed of the order 4 to 5 km/sec. Engines with a nuclear reactor can generate a gas discharge speed that is greater by a factor of 2, an electrothermal engine by a factor of 4-5, and ion and plasma engines greater by a factor of 10-100.

The requirement to produce and convert energy in the individual units limits the power which is communicated to the working substance. Therefore, the mass flux of working substance must be held at a comparatively low level. This constraint leads to small values of thrust and acceleration being typical for electrical jet engines. However, such accelerations can be maintained for a long period, so that vehicles with these engines can reach large final speed, up to 200 km/sec. /124

Small accelerations created by electric engines preclude their use if the flight takes place in a region where there is an appreciable aerodynamic drag or where gravitational forces have to be overcome to make a trajectory change. But they can be used successfully where there is no resistance to motion.

Electric engines can be of several types.

The plasma engine. A plasma-electric engine is a combination electrothermal engine system. Its thrust is generated by expanding a hot plasma. The source of energy for operation of this engine is an electrical generator. A schematic diagram of its construction is shown in Figure 53.

The engine consists of a cylindrical chamber with electrodes at each end. The negative electrode (cathode) is a nozzle. The positive electrode (anode) is a special material in the form of a rod. During start-up and operation of the engine, direct current is supplied to the electrodes. At a certain potential difference, an electric arc forms. The electric energy in the arc is converted to ionization of the atoms and dissociation of the molecules, and also to increased temperature in the space surrounding the arc. The positive electrode has an automatic feed to supply the material consumed in the arc. Ionized atoms /125
and dissociated molecules form the plasma. The plasma creates

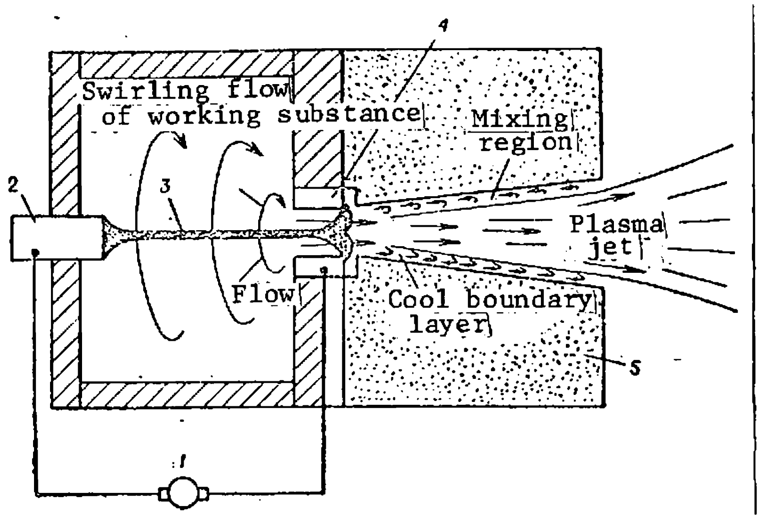

Figure 53. Schematic of plasma-type electric jet engine.
1- D C generator; 2- anode; 3- electric arc; 4- cathode;
5- nozzle tank.

the thrust in discharging from the nozzle.

To create a plasma engine operating on this principle, cooling is required and the structural elements must be protected from erosion. The main defect of such an engine is the need to use a comparatively large amount of working substance (coolant) in space flight. In addition, the hot plasma acting on the nozzle leads to fast erosion of the structural elements with which it makes contact, if protective measures are not taken.

In its operating principle, this kind of plasma engine is very close to a thermal engine. Therefore its specific impulse, although higher than for ordinary liquid fuel engines, is still limited to relatively small values.

To obtain electrical energy in plasma engines requires cumbersome equipment to convert the primary energy into electrical,

which considerably increases the mass of the total power unit.

The <u>ion engine</u>. This engine is also called an electrostatic engine. In it particles of the working substance having an electrical charge, are accelerated in electrostatic fields. The discharging particles can be ions, charged particles, or even dust particles and droplets. The discharge speed of particles leaving the accelerating chamber is determined by the potential difference at the ends of the chamber, and the particle charge and mass. The conversion of electrical energy into kinetic energy of the discharging beam of particles is described quite simply in the theory of electrostatic engines. However, to create ion engines a designer is faced with very complex problems.

In order to maintain electrical neutrality of a spacecraft body, an ion engine must discharge strictly equal amounts of positively and negatively charged particles. The most accurate method for this condition to be fulfilled is simultaneous discharge of positive ions * and electrons, which are liberated in the ionization process.

Figure 54 shows a diagramatic representation of an ion engine. The basic elements are a tank for storing the working substance, a system for feeding the working substance, a heater, an ionizer, an accelerating chamber, an electron omitter, and a neutralizer. The supply of working substance (cesium, mercury, etc.) from the tank is achieved by displacement. In the simplest case, the tank is a metal bellow in which the working substance is stored in the liquid state and is discharged in the amount required to feed the engine. /126

* An ion is a charged atom (or group of atoms); it differs from a normal neutral atom in having an excess or a deficiency of one or several electrons.

The liquid working substance passes into the heater where it is vaporized. In the gaseous phase, the working substance reaches the ionizer 1. Here it is converted into positively charged ions and negatively charged electrons. These particles arrive in the accelerating chamber. There an electric field is formed by the electric current which passes through the electrodes 3. Forces acting in the electric field accelerate the particles to very large discharge speeds, 100 km/sec and more. Then the ions and the electrodes are again mixed due to the action of the neutralizer 4, and form a neutral plasma.

In the last 30 years a number of theoretical and experimental programs have been conducted to determine the potential capability and the advantages of electrical rocket engines for carrying out many space missions. However, the practical application of these engines is limited by the fact that they can be used only for flights designed with a long-duration continuous operation of the engine (e.g., for significant changes in orbital parameters or in interplanetary flights with large payloads), which cannot be achieved on technical grounds, in particular, because of the limited resources. Another no less important factor limiting the use of these engines is that they require energy units with large capacity and low specific weight. Therefore, all space flights accomplished in the sixties used rocket motors with chemical fuel. In the seventies, the space program plans, particularly in the USA, provided for many tasks which could be effectively performed only using electrical engine units. As has been mentioned, in chemical rocket motors the maximum discharge speeds are 3-5 km/sec, and in nuclear rock engines 8-9 km/sec. The acceleration of charged particles by means of electrostatic or electromagnetic fields would yield discharge speeds exceeding these by several orders of magnitude. However, in defining areas for logical use of spacecraft carrying

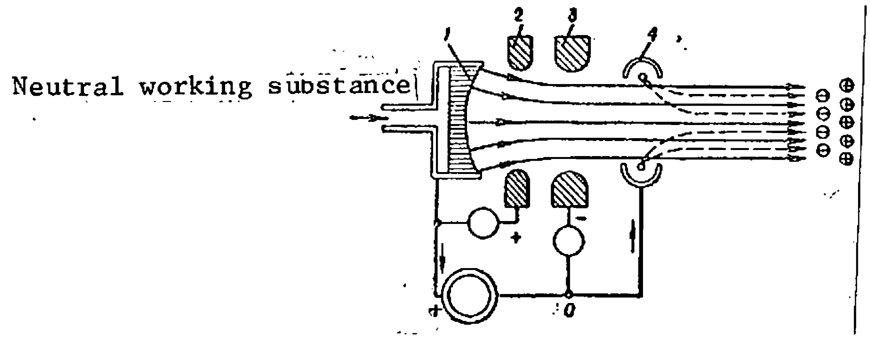

Figure 54. Schematic of ion engine.
1- ionizer; 2- shaping electrode; 3- accelerating electrode; 4- neutralizer.

these engines it has been established that an electrical rocket motor does not need to have the maximum possible discharge speed. /127

For each specific task the optimum discharge speed for which a rocket with an electric engine gives the maximum relative payload is determined as a supplement.

It has also been established that rockets with electric engines are particularly suitable for long duration space flight (at least more than a month). The optimum discharge speed fall in the range from 20 to 200 km/sec for a wide class of space missions, from near Earth to interplanetary flight.

Electric engines designed for a broad class of tasks during space missions have advantages in comparison with rocket motors operating with chemical fuel.

It is reported in the foreign literature that ion engines have been tested in space in the USA, while work on other types of electrical engines is in the investigative stage.

The space programs in the USA for the period 1970-1985 envision the use of electric engines with electrical power derived mainly from solar installations. Plans have been developed for manned space vehicles with electric engines for flights to Mars. Since the radioisotope units which have been constructed and developed are designed for low power levels, while work in the area of high-power reactor units is in the stage of preliminary investigation, the present studies are based on solar electrical supplies, as being the most developed and having the most suitable characteristics. An important factor which would enable solar energy sources to be used for electric engines is that they have been developed with low specific mass (10-15 kg/kW). Because the electric engines developed at present have low power, auxiliary electric engines of power 10 kW and more for long-duration interplanetary flight, will take the form of a combination of medium-power engines.

Important factors in evaluating the capabilities of such systems are the reliability and the service lifetime of the engine. Because of the considerable erosion of their elements, plasma engines have a lifetime of several hundred hours, while individual ion engines have operated in life tests up to 10,000 hours. However, the problem of achieving reliable operation during a long flight of a system consisting of several electric engines, is complicated by the fact that the validity of reliability factors for a compound system can be established with a high degree of accuracy only by long-term and expensive testing. Therefore, it is necessary to find methods for determining the increase in the general reliability of the system when /128 the reliability of components is not high enough or has not been accurately determined. Under these conditions it is usual to employ duplication of units of the system. However, duplication always leads to an increase in the system mass. As regards

space engine systems there is a requirement to minimize the increase in mass resulting from duplication.

An electric engine system is conventionally divided into the following subsystems: the electric power source, the fuel supply subsystem (working substance), and the engine. Foreign engineers have analyzed the reliability of such a system, relevant to an electric engine with solar energy unit. The reliability can be increased considerably by using enough duplicated elements. Since the engine unit can be designed in the form of mutually interchangeable units or modules, it is desirable to find a combination of operational and duplicated modules which will give minimum mass for a certain total reliability of the system. In addition, for some interplanetary flights, for example to Mars, the energy obtained from solar batteries will diminish as the distance from the Sun increases, and therefore in certain periods of the flight, it will be necessary to switch on specific modules of the engine systems and modules of the energy source systems. If the modules are constructed so that they can be renovated and again switched in following a malfunction, then all the modules in use can be considered as spares.

For motion in an interplanetary trajectory in which the distance between the spacecraft and the Sun varies with time, the engine and the solar power source must be matched to operating conditions of constantly changing parameters at the output of the solar cell, since, because of the variation of the distance to the Sun, the intensity of the light energy is varying, and therefore, so is the voltage in the solar panel. As the spacecraft approaches, the target planet, some of the modules of the engine system must be switched on, since there is a decrease in the power captured by the solar cell. Since it is required that the engine operates with constant voltage, control of the voltage within certain limits may be necessary.

In a modular power system, the source consists of a series-connected energy module of low power and low voltage. The output voltage of the electrical sources of this kind can be controlled by switching some of them in and out.

It is known that an energy system for this kind of electric engine consists of a large number of electrical energy sources for different elements of the engine. The main power is the source for acceleration of the ion beam. It is a high-voltage source, which controls the voltage to maintain a constant specific impulse of the engine. Analysis of an engine of power 48 kW, meeting the requirements of maximum reliability, /129 as well as the requirement for packaging on a spacecraft, has shown that the engine system consists of ten thrust modules and eight mercury reservoirs. Initially eight engines operate and two are spare. The engine system includes thrust modules using electron bombardment and oxide cathodes, having a diameter of 50 cm, a power of 6 kW, a current density of 2-3 mA/cm^2, and capable of a long lifetime.

The main special feature of the fuel supply system is the presence of a valve on each of the eight liquid mercury reservoirs. For this reason, any number of (the eight) mercury tanks can be used simultaneously to supply fuel in any quantity to any number of the eight operating engines. The use of several tanks increases the reliability of fuel storage, since if one tank goes out of action, it can be excluded from the engine system.

The energy unit consists of eight energy panel modules, and any one module is enough to supply the electrical energy for one thrust module.

CHAPTER 6

HISTORY OF RUSSIAN ASTRONAUTICS

THE FIRST ROCKET RESEARCH

In Russia, as in other countries, the use of gunpowder rockets (used first for war purposes) goes back to the nineteenth century. Russian scientists made a great contribution to the development of rockets. For example, in 1814, A. D. Zasyad'ko began to build military rockets; in 1849, I. I. Tretesskiy suggested the idea of using a jet of gas or vapor to power lighter-than-air machines; in 1861, the work of K. I. Konstantinov on military rockets was published; in 1866, M. N. Sokovnin suggested a scheme for a rocket-powered balloon; in 1880, the inventor S. S. Nezhdanovskii conceived the possibility of constructing a jet aircraft, powered by the energy of an explosive mixture: kerosine and nitric acid; in 1881, the revolutionary N. I. Kibal'chich, while imprisoned for his part in an attempted assassination of the Tsar, developed a plan for a device floating in air, operating on the rocket principle.

The basis for astronautics was laid down by K. E. Tsiolkovskiy. In 1883, in the publication "Free Space", he was the first to describe a spacecraft with a rocket motor. In 1895, in the publication "Visions Concerning the Planet Earth", he foreshadowed the possibility of creating an artificial Earth satellite. In 1903, Tsiolkovskii's classic work was published - "Investigation of Outer Space by Means of Rocket Devices". In this publication he gave a detailed description of a rocket device and its liquid motor, and put forward the fundamentals of the mathematical theory of rocket flight. In 1912 he proposed the

idea of an electric rocket motor, in which the discharge products were charged particles.

The ideas of Tsiolkovskii opened the door for man to reach into space. Academician S. P. Korolev has said that time inexorably erases what has gone before, but the ideas and work of Konstantin Eduardovich Tsiolkovskii will continue to attract increasing attention as rocket technology develops. He was a great and genuine scientist who lived far ahead of his time.

The fundamentals of the theory of rocket motion pertaining to spacecraft were developed by N. E. Zhukovskii in the papers "Reactions Involving Efflux and Influx of a Fluid" and "The Theory of Ships, Set in Motion by the Reactive Force of Water". / 131
The Russian scientist, I. V. Meshcherskii in the years 1893, 1897, 1904 and 1918 published theoretical papers which presented the basic equations of rocket dynamics.

The Great October Socialist Revolution opened up great horizons for the enthusiasts of rocket technology.

In 1929 Yu. V. Kondratyuk, who had begun the work earlier, published a rigorous mathematical account of results of his investigations into rocket and space problems.

One of the Russian contributors to the theory of rocket motor design, F. A. Tsander, in the book "Problems of Flights Using Jet Devices", published in 1932, summarized ten years of his work on topics in astronautics. In 1930 he constructed the first Russian laboratory motor OR-1 (test jet) of a new type. The motor operated on compressed air and benzine. Later on, Tsander took part in the development of liquid rocket motors, operating on liquid oxygen and benzine.

In April, 1924, at the Military Science Unit of the Air Force Academy, a Section dealing with interplanetary communication was created, which was to play a large role in popularizing the ideas of astronautics. In 1925 in Kiev, under the leadership of Academician D. A. Grave, a group was formed to study and attack problems of space flight, and this group arranged an exhibition dealing with space flight problems.

In 1928, at the Leningrad Institute for Communication Engineering, a section for its interplanetary communication was set up under the leadership of N. A. Rynin. Rynin, whose lectures on aerodynamics were attended by the author of the present book in the Military Science School in 1922, was the author of an Encyclopedia of Interplanetary Information, unique in its way, published in 1928-1932. It was published in 1929 with the objective of setting up a national or international institute for interplanetary information.

On the initiative of B. S. Petropevlovskiy, in Leningrad, in the Gasdynamics Laboratory (GDL), which came under the Wartime Scientific Research Committee of the Revolutionary War Council of the Republic, V. P. Glushko began work to manufacture Russian liquid rocket motors. In this laboratory, founded in 1921, experimental liquid rocket motors were developed during these decades. Simultaneously, solid-fuel motors were developed in the GDL, and were used as take-off accelerators for aircraft of different types, from the light aircraft of the U-1 type to heavy bombers of the TB-1 type.

In 1931, at the Special Unit for Aviation and Chemistry, the Moscow and Leningrad Group for the Study of Jet Propulsion was set up. Among those who took part in the Moscow group were: scientists and engineers: V. P. Vetchinkin, the scientist

N. E. Zhukovskiy and his very close colleague, S. A. Chaplygin, S. P. Korolev, F. A. Tsander, M. K. Tikhonravov, Yu. A. Popedonostsev, I. A. Merkulov, B. I. Cheranovskiy, and others, and those who took part in the Leningrad group included N. A. Rynin, V.V. Razumov, A. I. Shtern, I. I. Kulagin, E. E. Chertovskiy, and many others. /132

Shortly thereafter, a design and construction section was organized in the Moscow group whose task was to design and construct liquid rocket motors. The head of this organization was the engineer, S. P. Korolev, later an Academician, and a prominent designer of present-day rocket systems.

On August 17, 1933, the joint group launched the first Soviet liquid rocket, operating on liquid oxygen and gelatinous benzine. The same group also designed liquid rocket motors of Tsander type, also operating with two-component propellant.

At the end of 1933 the Scientific Research Institute was created and staffed by a creative group of Soviet rocket engineers, who built a number of experimental ballistic rockets and motors for them. In 1939 flight tests of a winged rocket were performed, and then ground tests for a rocket aircraft designed by S. P. Korolev. It was in this rocket aircraft with a liquid rocket motor that the pilot V. P. Fedorov made the first flight in 1940. In 1942, G. Ya. Bakhchivandzhi accomplished the first flight in a fighter aircraft with a liquid rocket motor, built under the leadership of V. F. Bolkhovitinov. The same institute developed the RS-82 and RS-132 rocket missiles, which were in use as early as May, 1939, in military exercises of the Russian fighter command during the hostilities in the Khalkhin-Gola region.

The Il-2 attack aircraft, designed by S. V. Il'yushin and equipped with such rocket missiles, were a powerful weapon during World War II.

Before the start of World War II, the BM-13 rocket launch unit was built, affectionately nicknamed the "rocket truck" (katyusha) in this country. This formidable weapon was first used in 1941 against the German forces.

The history of Russian rocket development is closely associated with groups of early enthusiasts of this new form of technology, which has been, and continues to be, very important in the development of a number of major scientific and engineering tasks, directed at the construction of powerful multistage launch-vehicles, rocket motors, and systems of automatic control for space vehicles.

UNMANNED SPACECRAFT

During the thirties in the Soviet Union, in parallel with the construction and application of solid rockets, liquid rocket motors began to be used for meteorological investigation of the upper atmospheric layers.

Successful tests of the application of different motors to rockets, culminating with the launch in 1939 of a two-stage rocket (the results of the tests were used in 1940 in the construction of aircraft and in subsequent work to improve them), proved that it was actually possible to design and build large rockets with greater flight range. /134

Successful testing of intercontinental ballistic rockets showed that flight in outer space was technically feasible and that artificial Earth satellites could be launched.

To Aleksander Nikolayevich Panomarev, with respect.

In this regard, the work of S. P. Korolev was of great importance. An eminent scientist and investigator and a talented organizer, he was the main designer of contemporary rocket systems. His name is associated with the design, the production management, and the operational use of powerful rockets which achieved the historical firsts of putting an artificial Earth satellite in orbit, and planting the Soviet penant on the Moon, and they were also used to photograph the side of the Moon invisible from Earth from an unmanned spacecraft. Under his leadership, manned spacecraft were designed and built which made the first manned flight in space and put man into outer space, and he directed the launch, flight, and landing of these spacecraft.

In 1911 Tsiolkovskiy wrote that man's first great step would be to fly beyond the atmosphere and create a satellite of the Earth.

Forty-six years later, this step was taken in the country of that eminent scientist. On October 4, 1957, the first artificial Earth satellite was launched in the USSR. This new heavenly body, appearing in the Solar System on that day, caught the interest of the whole world. The satellite flew above the terrestrial globe in an elliptical orbit, distant 947 km from the surface of the Earth at apogee (the most remote point of the orbit from Earth), and 228 km at perigee (the nearest point of the orbit). The satellite could be seen from the surface of the earth during the hours of evening, night and morning, even with the unaided eye. It was a bright star, moving amongst the fixed stars. The design of the first artificial Earth satellite seemed outwardly very simple. It was a silver-grey aluminum sphere with four whisker-like stub antennas. The radio equipment and the power supplies were contained inside its case. The temperature inside the satellite was determined mainly by the intensity of

the solar radiation. Its mass was 83.6 kg. During its lifetime (three months) the satellite completed about 1400 revolutions around the Earth and was able to verify a number of scientific and engineering predictions, and investigate the transmission of radio waves through the ionosphere. The deceleration of the satellite gave information on the atmospheric density at heights which were previously unattainable for an aircraft.

The satellite enabled studies to be made of the properties of the upper layers of the atmosphere and observations on processes /135 taking place on the satellite during its motion around the Earth, in particular its temperature and electrical charge.

The second artificial Earth satellite was placed in orbit on November 3, 1957, and it was the first satellite to carry a living creature. For the first time ever a living creature had escaped from the planet Earth. The second satellite reached a height above the Earth almost twice as great as the first, and its mass was six times larger than that of the first. The dog, Laika, was placed in a special compartment of the satellite. Laika's compartment was equipped with a life-support system, and a supply of water and food. In order to observe the main physiological processes, the animal was surrounded by instruments whose readings were transmitted to Earth by radio-telemetry. The successful flight of Laika dispelled many doubts concerning the possibility of a comparatively long period of residence of a living organism in conditions of outer space.

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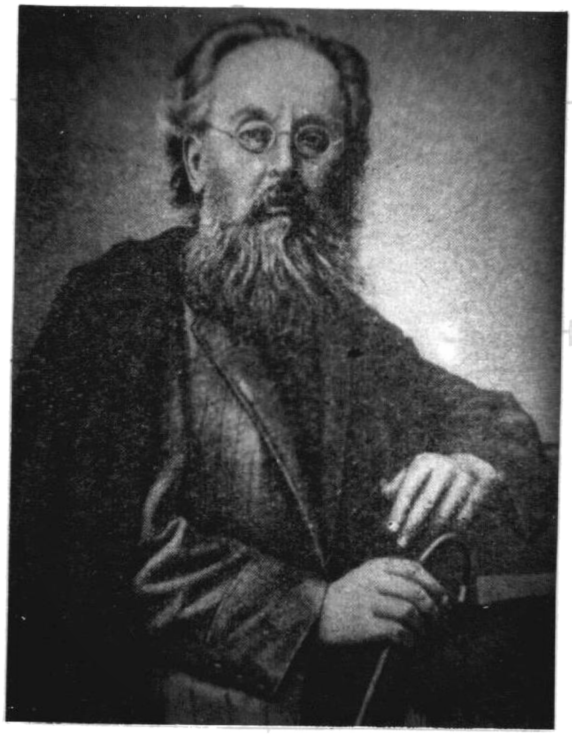
The satellite carried instruments to investigate solar radiation in the ultraviolet and x-ray spectral regions, instruments to record cosmic ray particles, and a telemetry system to transmit to Earth the information gathered by the scientific equipment.



Konstantin Eduardovich Tsiolkovskiy

The third artificial Earth satellite was an automatic laboratory, huge for its time, which was launched on May 15, 1958. Its scientific, measuring, and radio-telemetry equipment allowed various experiments to be conducted, and the results stored and transmitted to Earth.

Continuing their program of space investigation, Soviet scientific organizations began the systematic launch of the Cosmos series of artificial Earth satellites in 1962, which accomplished a large program of scientific investigations. They conducted investigations of solar activity, determined the state of the atmosphere, including the ionosphere, gathered important

meteorological information, made a magnetic survey, and conducted medical and biological investigations.

For example, one of the satellites, Cosmos 7, was designed to investigate the radiation arising after a nuclear explosion, a study which was required to ensure radiation safety for manned flights of the Vostok 3 and Vostok 4 spacecraft in August 1962. The Cosmos 110 satellite was launched into the radiation belt surrounding the Earth with test dogs, and was returned to Earth successfully. The main objective of the experiment was a medical and biological investigation of the long-term effect (period of 22 days) of weightlessness and other space flight factors on the organs of animals.

The investigations conducted by satellites of the Cosmos series helped to solve technical questions facing astronautical science. For example, Cosmos 186 and Cosmos 188 were the first, in October, 1967, to carry out automatic homing, rendezvous, docking, joint flight and undocking in space. This brilliant experiment was later repeated successfully by the satellites Cosmos 212 and Cosmos 213. The flight confirmed that orbiting space stations could be built by being welded together in situ, and could be serviced by means of spacecraft which would maintain a link with Earth. Figure 55 shows docking of the spacecraft Soyuz.

The versatile spacecraft Polyet 1 and Polyet 2 were launched on November 1, 1963 and April 12, 1964, their objective being to develop a system for maneuvering and flight stabilization for spacecraft, this being necessary for rendezvous and docking in space. The experiment yielded data which were used to design new systems of flight control for spacecraft, and also data on the assembly of spacecraft in Earth orbit, and on the construction of permanent orbital stations.

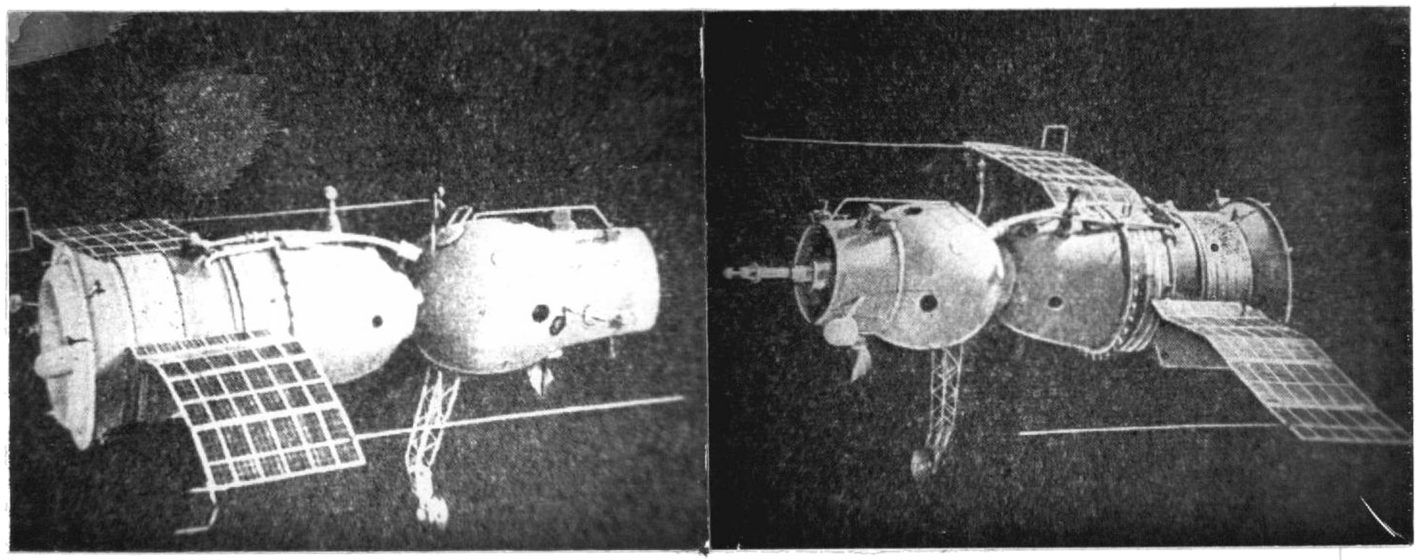


Figure 55. Automatic docking of the Soyuz spacecraft.

C - 3

Two artificial Earth satellites were launched on January 30, /138
1964, the scientific stations Elektron 1 and Elektron 2 - their objective being a study of conditions for radiation safety of space flight.

These stations were launched in different orbits, distant 7000 and 68,000 kilometers from the Earth, respectively: later the experiment was repeated with the Elektron 3 and Elektron 4 spacecraft. The launch of the two Elektron satellite systems enabled a continuous investigation to be carried on for almost a year of the intensity of radiation at different heights and a dosimetric chart of the Earth's radiation belts to be constructed. The data obtained have been very important in ensuring radiation safety for manned space flight.

In 1965, the massive orbiting spacecraft Proton 1 and Proton 2 were launched (each had a mass of about 12 tons), and in 1968, Proton 4, with mass about 17 tons, was launched — this being the largest of the orbiting scientific spacecraft at that time. These spacecraft were launched to study the nature of primary cosmic ray particles of high and very high energy.

The successful launches of these early artificial Earth satellites have shown that the Russian rocket designers have mastered flight in near-Earth space at velocities close to the first cosmic speed. The next problem that Soviet astronautics faced was to reach flight at speed close to the second cosmic speed, and to reach out to investigate the space around the Moon.

It is well known that Galileo was the first to direct his telescope towards the Moon in 1610. However, in the course of more than three and one-half centuries, only one-half of the lunar surface, that constantly turned towards the Earth, was available for observation. The other half was hidden from the human eye, and therefore only a map of the lunar surface visible from Earth existed. The flight of the Russian spacecraft Luna 1, launched on January 2, 1959 (Figure 56), ushered in a new era in lunar studies.

The last stage of the rocket, having traversed the Earth's atmosphere, was the first object to reach the second cosmic speed, about 11.2 km/sec, and to be placed on a course towards the Moon. The spacecraft was equipped with an automatic laboratory to investigate space. The spacecraft had a mass of 361.3 kilograms. The main objective of the launch was to reconnoitre the path to the Moon. On January 4 it passed near the Moon at a distance of 5,000 - 6,000 km from the surface and became the first artificial planet of the solar system. It is now flying in space between the orbits of Earth and Mars, making one revolution around the Sun in 450 days.

The second lunar spacecraft, Luna 2, was launched on Septem- /139 ber 12, 1959. It carried a pennant with a picture of the official crest of the Soviet Union. On September 14 it made a hard landing on the lunar surface. The first object to leave the Earth had landed on the Moon.

During the flight of Luna 3, launched on October 4, 1959, the side of the Moon invisible from Earth was photographed. This was the first successful experiment in the history of mankind to photograph and transmit pictures of another heavenly body from space to Earth. Two cameras carried on Luna 3 photographed the side of the Moon invisible from Earth, except for a segment of the surface outside the field of view of the camera objectives.

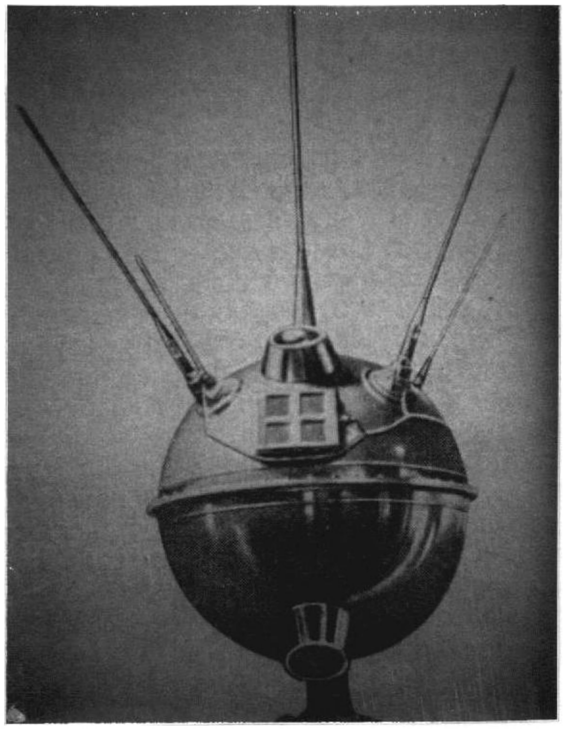


Figure 56. The Luna 1 spacecraft.

This part of the lunar surface was photographed by the unmanned spacecraft Zond 3, whose flight began in July, 1965, and ended in March, 1966. Its objective was not only to photograph the Moon, but also to test onboard equipment under conditions of long-duration space flight.

At the end of its flight, Zond 3 was distant 150 million kilometers from Earth, equal to the distance between the Earth and the Sun. / 140

The building of multi-stage rockets made it possible to launch spacecraft to the Moon, not only from the Earth's surface, but also from the orbit of an artificial satellite, which made it possible to continue studies of the Moon using unmanned spacecraft.

Other Moon probes included Luna 4, Luna 5, Luna 6, Luna 7, and Luna 8, which continued the exploration of the Moon and were intended to solve the problems of a soft landing of a capsule with scientific equipment on the lunar surface, development of space attitude control devices, control of onboard equipment, and radio control of trajectories.

On February 3, 1966, a soft landing of the unmanned spacecraft Luna 9 on the lunar surface was accomplished. The mass of this spacecraft at injection into its lunar trajectory was 1583 kilograms. It was launched from Earth orbit and, in addition to scientific equipment, had a special engine for trajectory correction and deceleration in the lunar flight, as well as a control module. After making appropriate corrections on radio command from the Earth in its approach to the lunar surface, the motor was switched on at a distance of 75 km automatically, and Luna 9 (Figure 57), which had previously separated from its retro-engine, made a smooth landing on the lunar surface.

It carried out seven radio-communication sessions with Earth and transmitted a telemetry picture of a panorama of the lunar surface. The spacecraft systems operated on the surface of the Moon for 75 hours.

The flight of Luna 9 showed that it is possible to accomplish a soft landing of spacecraft on the lunar surface.

Luna 13 (Figure 58), launched from Earth on December 21, 1966, had the same television system as Luna 9 for transmitting pictures of the lunar landscape to Earth. Besides the telemetry from the Moon, Luna 13 investigated the top layer of the lunar soil. It turned out to have a density of 0.8 g/cm^3. i.e., less than the average density of the Moon and considerably less than the density of Earth soil.

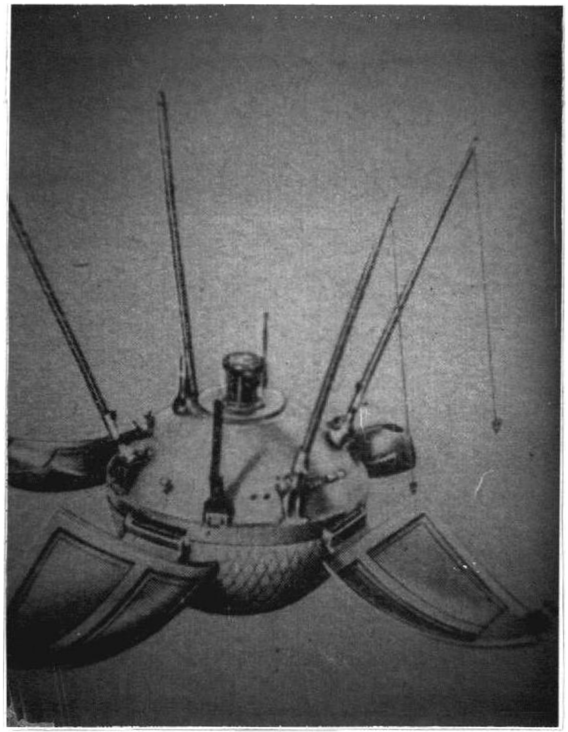


Figure 57. The Luna 9 spacecraft.

Other stations (Luna 10, Luna 11, Luna 12, Luna 14, and Luna 15) investigated space near the Moon, studied the meteor flux, the radiation environment and the magnetic field, and transmitted the information to Earth. In addition, they photographed the lunar surface.

During the course of the XXIII Session of the Communist Party of the Soviet Union, the delegates and guests listened with profound emotion to the party anthem,"The International",transmitted from Luna 10, which was the first artificial satellite of the Moon. The operating period of the first artificial lunar satellite lasted for 56 days. During this time it made 460 orbits around the Moon, and transmitted a large volume of scientific information to Earth.

Scientists obtained much new information concerning the gravitational and magnetic fields of the Moon, the chemical composition and radioactivity of its soil, and the magnetic wake of the Earth, through which the Moon and its satellite passed periodically. /141

The sixties were characterized not only by investigations of the Moon and its surrounding space. Investigations were also made of more remote heavenly bodies - of Venus and Mars in particular.

Venus is one of the most mysterious planets of the solar system. It is always covered by an opaque layer of thick cloud, which hides its surface from observers on the Earth. Therefore, there was interest by scientists everywhere in using spacecraft to study it. In its motion in orbit, Venus periodically passes through two oppositions relative to the Sun and the Earth. Then the minimum distance between Venus and Earth is about 42 million kilometers, and the maximum distance 258 million kilometers. / 142
Therefore, the first task of our scientists was to determine a suitable interplanetary trajectory for a flight to Venus.

On February 12, 1961, the first Soviet spacecraft Venera 1 was launched towards Venus. Its mass was 643.5 kg. It was launched from a heavy artificial Earth satellite. Venera 1 accomplished the majority of its mission tasks. Communication with the spacecraft was continued out to a distance of 23 million kilometers. Venera 2, and later Venera 3, were launched in November, 1965. On March 1, 1966, Venera 3 reached the surface of Venus and planted on it the first pennant of the Soviet state. It had traversed about 350 million kilometers on its interplanetary path.

Later the spacecraft Venera 4, Venera 5, and Venera 6 were launched, which transmitted to Earth data on the atmosphere of the

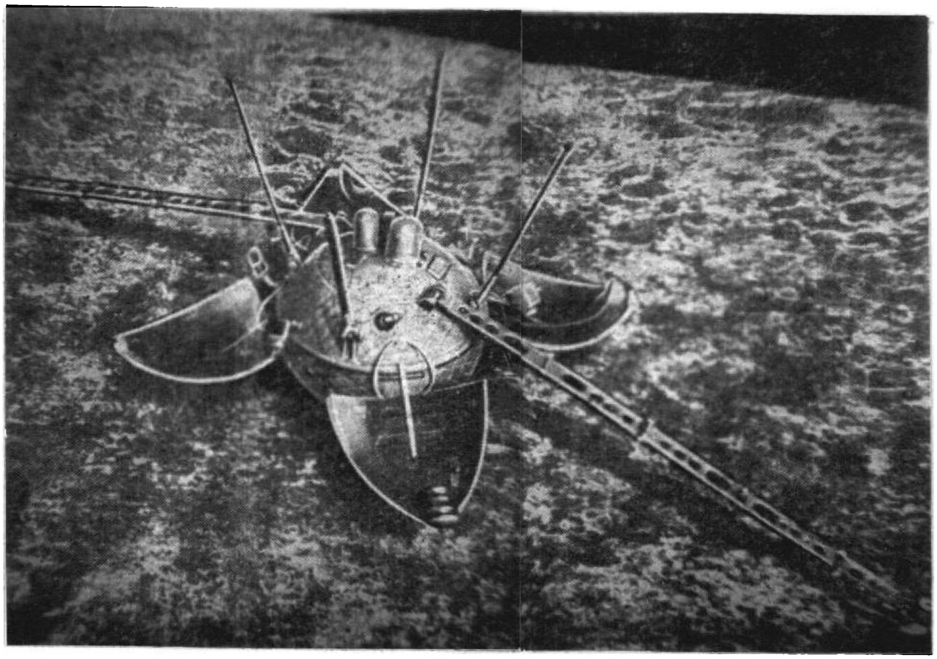


Figure 58. The Luna 13 spacecraft (investigation of lunar soil).

planet. It was established that 90% of the atmosphere was made up of carbon dioxide. The flights of Venera 5 and Venera 6 and the smooth descent of their entry capsules into the atmosphere of Venus allowed telemetry information to be transmitted from each stage to Earth during sessions lasting for 53 and 51 minutes, respectively. Data were obtained on the structure of the plasma flux surrounding the planet, on ultraviolet radiation, and on the content of carbon dioxide, oxygen, water and nitrogen in the atmosphere at different levels above the planet's surface. The exploration was made in the height range 25-40 km. /143

The Soviet program in space exploration also included launches of the spacecraft coming under the general classification of Zond. The function of these spacecraft was development of certain systems and components of future spacecraft, the conduct of scientific investigations, and the improvement of methods of navigation in deep space. In particular, Zond 2 was the first spacecraft to

Sergey Pavlovich Korolev

test plasma rocket motors, for trajectory correction purposes.

Some especially important problems were addressed on Zond 5 which, after completing an Earth-Moon-Earth flight, was the first to reenter the Earth's atmosphere with the second cosmic speed, and accomplish a ballistic entry, landing in an assigned region of the Indian Ocean. /144

The Zond 6 spacecraft, which flew around the Moon and took photographs, returned tortoises to Earth which were carried on the flight. Before atmospheric reentry, the descent capsule was separated from the spacecraft. Its aerodynamic shape and reliable heat shield allowed it to make both a ballistic and a controlled descent into the Earth's atmosphere.

As was mentioned earlier, the considerable loads and heat fluxes acting on a spacecraft during atmospheric entry at a speed of about 11.2 km/sec are significantly reduced during a controlled aerodynamic descent. In this case, the entry capsule, having first penetrated into the atmosphere, flies over a section of the Earth and soars upwards because of its lift force. However, the force of Earth gravity draws it back again into the atmosphere, but now with much less speed, about 7 km/sec. In this way, due only to passive deceleration in the atmosphere, the speed of motion of the descent capsule is reduced by almost a factor of 50, which allows a parachute to be deployed. In particular, Zond 7, after flying around the Moon, made a soft landing on the Earth after a double entry of the capsule into the atmosphere (skip-up and reentry), and because of aerodynamic drag, its speed was reduced to 200 m/sec. These experiments were important in ensuring safety of human flight on interplanetary journeys.

On November 1, 1962 the unmanned spacecraft Mars 1 was launched towards Mars. For a period of almost five months, stable radio-communication was maintained with the spacecraft. In this time the spacecraft withdrew from Earth to a distance of about 106 million kilometers. The spacecraft gave clear transmissions of the data concerning the environmental conditions in deep space between the orbits of Mars and Venus, and transmitted data on the previously unknown meteor flux.

An important step towards flight in near space were the launches of the Soviet spacecraft-satellites, on May 15, 1960. The main tasks of the spacecraft launched on August 19, 1960 was to develop systems providing normal life-support for man in a sealed cabin and a safe flight of a spacecraft and reentry to Earth. The spacecraft cabin contained a compartment with test animals, the dogs Belka and Strelka. A separate section of

the compartment contained 12 mice, insects, plants, fungus cultures, seeds of grain, peas, maize, onions, and other biological material. The presence of living creatures in the cabin required accurate observation of temperature conditions, regular supply of food and drinking water, removal of carbon dioxide, strict dosimetry in the supply of oxygen as a component of the air mixture, and the development of devices for eliminating wastes.

In accordance with the USSR space research program, three additional experimental spacecraft flights were made. The accumulated favorable experience in launches of artificial Earth /145 satellites and spacecraft made it possible for Soviet scientists to go on in real earnest to practical accomplishment of flight of manned spacecraft into deep space.

THE FIRST MANNED FLIGHT INTO SPACE

Early in 1961 construction was completed of the very first manned spacecraft, the Vostok. This was the culmination of the large and purposeful program of research and development work by the scientists.

The mass of the spacecraft was 4.73 tons. Structurally the Vostok consisted of two compartments: the entry spacecraft, and the instrument compartment. They were joined together by means of four tie-plates, released by means of an explosive lock. The descent capsule, which included the astronaut's cabin, took the form of a sphere of diameter 2.3 m, covered with heat-shield material. Its mass was 2.4 tons. The cabin had three windows, protected by heat-resistant glass, and also an ejection seat for the astronaut, dressed in his spacesuit (Figure 59). The seat included a system for spacesuit ventilation, parachute systems, and emergency ground supplies. The seat served simultaneously

Figure 59. View of the inside the cabin of the Vostok spacecraft.

as a method of escape in the event of a launch emergency. Special rocket motors attached to it could remove the astronaut from the danger zone and raise him to a height sufficient for reliable deployment of the emergency parachute system. To maintain the spacecraft in the correct attitude during flight, a one-axis attitude-control system was provided, using the Sun as reference. It consisted of gyroscopic and optical sensors, logic devices, and microjet motors, operating on compressed gas. The attitude control could be operated both automatically and manually. /146

For control of the spacecraft, the cabin had a pilot console with an instrument desk and a stick with a control unit. Using a miniature globe of the Earth, located in the instrument desk, the astronaut could determine the position of the spacecraft relative to the Earth.

An air conditioning system in the cabin maintained normal pressure (about 750 torr), humidity, oxygen concentration and temperature in the range 12-25°C. Supplies of food, water, and

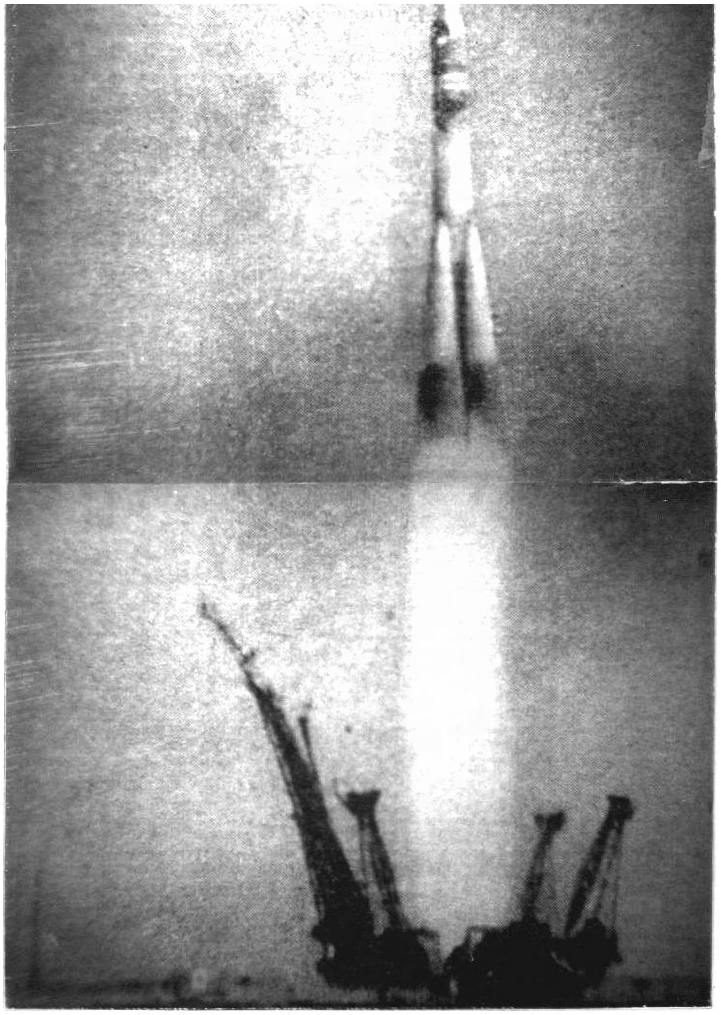


Figure 60. Start of the flight of the Voskhod spacecraft.

chemical materials for air regeneration were calculated on the basis of a ten-day flight. Vostok was equipped with radio equipment for communication with Earth, a telemetry system which made it possible for constant monitoring of the condition and state of health of the astronaut, the operation of the onboard systems, and the parameters within the astronaut's cabin, as well as television equipment for visual observation of the astronaut's condition.

On April 12, 1961 the first Vostok spacecraft, piloted by the astronaut Yu. A. Gagarin (1934-1968), was injected into artificial Earth orbit. The total duration of the flight from launch to landing was 1 hour 48 minutes (108 minutes).

How did this historic flight go? The spacecraft was made ready for the flight, everyone left the launch area, the flight controllers took their places in a special bunker and established contact with the astronaut and the launch team. The service booms were withdrawn from the rocket, the first stage motors began to operate, gradually developing power up to millions of horsepower. The rocket, swaying slowly, moved upwards (Figure 60), climbed away, set its course, and became invisible to the human eye.

As the speed increases, so do the loads increase, giving the effect of increasing the astronaut's weight by a factor of 4-5. The first stage motors cease to operate, and the four side units separate from the center unit. The second stage rocket continues to operate, passing up through the atmosphere. Then the third stage is ignited, accelerating the spacecraft to the first cosmic speed, equal to 8000 m/sec, or 28,800 km/hr. The Vostok spacecraft with an astronaut from the USSR went into an artificial Earth orbit. The astronaut reported to Earth that he was injected into orbit, made the first entry in the onboard journal, and observed that the pen with which he had written remained suspended in space in the cabin. This was the phenomenon of weightlessness. Yurii Gagarin passed through day and night in the space of his hour and one-half flight, although these concepts are very arbitrary in space. The concepts of up and down and right and left are conven- /148
tional. Even when flying in orbit, a spacecraft is constantly rotating slowly. Through a single window, the astronaut can see in succession the Earth, the Moon, the bright Sun, and the darkness of space. He does not sense the rotation, and does not feel that there is an up or a down.

The program for the first manned flight in space was accomplished, and the spacecraft must return to Earth. To do this, the spacecraft speed must be reduced and it must be set on a descent trajectory by switching on a retromotor. But before doing this, the rotation of the spacecraft must be stopped, since one must eliminate the possibility that the spacecraft obtain an additional velocity and transfer to a higher orbit, instead of a descent trajectory. The reaction jet of the retromotor is directed opposite to the direction of motion of the spacecraft, and therefore it decreases the speed and transfers to a descent trajectory.

At the end of the retro-burn, and upon transfer of the spacecraft to a descent trajectory, the instrument compartment is separated, and the descent capsule enters the atmosphere, where its speed is reduced because of aerodynamic deceleration. The temperature of the air surrounding the entry capsule reaches several thousands of degrees. The spacecraft flies on, as if embraced by flames, but the Vostok was covered with a thick layer of a special heat-shield material to maintain normal temperature in the astronaut's cabin; signs of scorching of the heat shield were observed after the landing. Figure 61 shows the spacecraft after its safe return from space.

Yurii Alekseevich Gagarin ushered in the era of manned flight in space. The first flight in history into space demonstrated that human flight in space was a practical possibility. The flight of Vostok seemed like the miracle of the century, but it was no miracle; it was an actual event. This flight was a victory for our scientists and engineers, and a clear manifestation of the creative genius of the whole Soviet nation.

"Ten Days Which Shook the World" was the title given by the American author John Reed, a good friend of the Soviet Union, to his book on the Great October Socialist Revolution. The "astonished world" entered into a new era in human history, the era of Communism.

The 108 minute flight of Yurii Gagarin again astonished the world. His flight was a great success for all who planned and built the spacecraft with all its systems, all who directed the flight, trained the astronauts, and reported the possibility of the flight of a Soviet citizen into space in a Russian manned spacecraft.

A total of only 40 years separated the launch of the first artificial Earth satellite from the Great October Socialist Revolution. The road to penetration of space began at the time when Vladimir Il'ich Lenin developed the GOELRO plan, when the Communists and Komsomol members of Volkhovstioya built the dams of the first hydroelectric power station during a period of fierce frost, when by superhuman effort of will and strength, the / 149
Soviet people created industrial centers, built new towns and factories, laying down the energy base of the country. Year by year the Communist party has led our country along Lenin's path, has educated a cadre of Soviet scientists and engineers, and has instilled a great love of knowledge in the people.

The conquest of space is one of the most difficult and complicated scientific and engineering problems of contemporary life.

Our land has become a pioneer in the conquest of space, and its scientists and engineers, workers and astronauts are the first to tread man's path into space. The Soviet Union has priority in reaching most of the milestones in astronautics:

Figure 61. A space flight is completed.

- launch of the first satellite (Sputnik 1) into orbit around the Earth, in 1957;

- launch of the first interplanetary spacecraft, Luna 1, in 1959;

- the first hard landing, that of Luna 2, on another heavenly body, in 1959;

- the first manned flight into space (Yu. A. Gagarin, in Vostok 1), in 1961;

- the first excursion of a human into space (A. A. Leonov, in Vokhod 2), in 1965. /150

- the first soft landing, of Luna 9, on the Moon, in 1966;

- launch into orbit of the first artificial satellite of the Moon (Luna 10), in 1966;

- flight around the Moon and return to Earth of the unmanned spacecraft, Zond 5, in 1968;

- building of the experimental manned spacecraft (Soyuz 4 and Soyuz 5), in 1969;

- building of the first orbital manned spacecraft Salyut, in 1971.

In honor of these world-shattering achievements of our country in the conquest of space, April 12 has become a worldwide day for recognition of aviation and astronautics. This day is solemnly celebrated by everyone, honoring the importance of Gagarin's flight for the development of astronautics and the great success of the Soviet Union in the study and conquest of cosmic space.

It is well known that in a flight in a spacecraft the loading rapidly increases as well as the speed. Can a human being survive this? Are there ground tests which can be done? How will the human body endure weightlessness? How will the descent from orbit go and the deployment of the parachutes? It became possible to answer these and other questions only after the first flight of a human being, a Soviet citizen, in space. The flight had a very great scientific and practical importance. It showed that a human can withstand conditions of space flight, injection into orbit, and return to the Earth's surface, meanwhile retaining his ability to coordinate movements and to think clearly.

Valentin Petrovich Glushko

It should be remembered that the first orbital flight accomplished by the American astronaut Glenn was made almost a year later, on February 20, 1962.

Academician Sergey Pavlovich Korolev has said of the first astronaut that he represents a happy combination of courage, analytical ability, and extreme dedication.

After his flight Gagarin continued to study assiduously and purposefully. He gave a brilliant defense of his diploma thesis, and graduated with distinction from the N. E. Zhukovskiy Air Force and Military Engineering Academy.

It is part of man's intrinsic nature to study all that is new and unknown. His thirst for knowledge concerning the

surrounding environment is unquenchable. This is his guarantee for the continuous unlimited progress of man. Our distinguished countryman, K. E. Tsiolkovskiy, stated fifty years ago that the planet Earth is the cradle of humanity, but man cannot always live in the cradle and man will not always remain on Earth; he is urged on towards the world and space, first penetrating beyond the atmosphere, and then conquering all of solar space.

The burgeoning development of science and engineering in recent decades was responsible for placing the first artificial Earth satellite into orbit in October, 1957, and for taking man's first step in 1961 into the unlimited space of the Universe. We have seen the start of man's space era; the conquest of space has begun, and a new profession has emerged, that of the astronaut.

FLIGHTS IN THE VOSTOK 2, VOSTOK 3 AND VOSTOK 4 SPACECRAFT

On August 6, 1961 the spacecraft Vostok 2, piloted by Major G. S. Titov (Figure 62) was successfully launched. The program for the second space flight, designed to carry out 17 orbits of the Earth, was aimed at performing a large number of scientific investigations and observations. Structurally the spacecraft did not differ essentially from the Vostok spacecraft.

The flight checked the capability of an astronaut under conditions of a long period in the cabin of a spacecraft in flight, the nature of the daily cycle of the life-support system on the human organism, and the special features of food intake. Periods of wakefulness and work for the astronaut alternated in a specific sequence with periods of rest and sleep.

After completing more than 17 orbits around the Earth and after flying 700 thousands of kilometers, the spacecraft returned

successfully to Earth. After the flight Titov said that it was very interesting to observe Earth from space: he could distinguish rivers and mountains, could easily see clouds and could distinguish them readily from snow because of the shadows which they threw on the Earth's surface.

The long period in conditions of weightlessness and the subsequent descent from orbit on an entry trajectory, accompanied by the effect of increasing loads, was not harmful to the astronaut's health.

The successful flight of almost 25 hours of Vostok 2 showed that the physical features of the space environment did not present an obstacle to long-duration flight of humans in space. On a group flight from August 11 to 15, 1962, on the spacecraft Vostok 3 and Vostok 4, A. G. Nikolaev and P. R. Popovich (Figure 63) verified the operation of onboard systems in conditions of a long period in space and the operation of a group of technical units, for spacecraft flying in the immediate vicinity of each other. This exercise formulated and improved systems for gound control of spacecraft simultaneously performing flights in near orbits. This flight was the first to carry out direct transmission of television pictures from a spacecraft to a network of relay stations for mass television, verified the possibility of astronauts remaining unrestrained in the cabin, and verified their operational activity in the unrestrained situation. The main /153
objective of the flight was to check that activity and operation were possible during a space flight extending over many days.

As Popovich has said, he was easily able to withstand the loads arising during injection of the spacecraft into orbit, and quickly adapting to weightlessness, he began to accomplish the planned program. To do this, he had first to unfasten himself from the harness system and float freely in the cabin. When

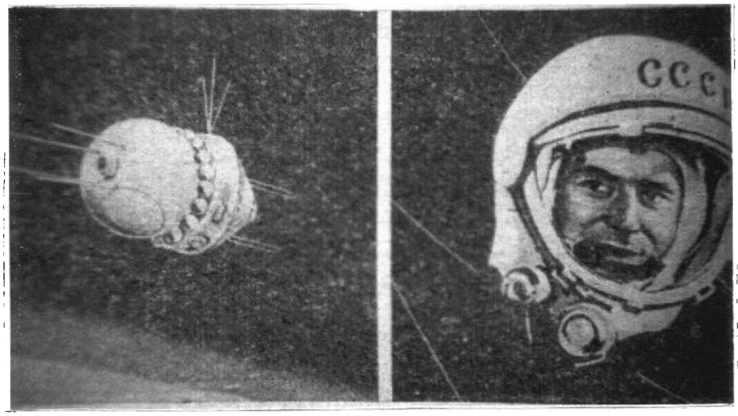


Figure 62. G. S. Titov, who completed a one-day flight in space in the Vostok 2 spacecraft.

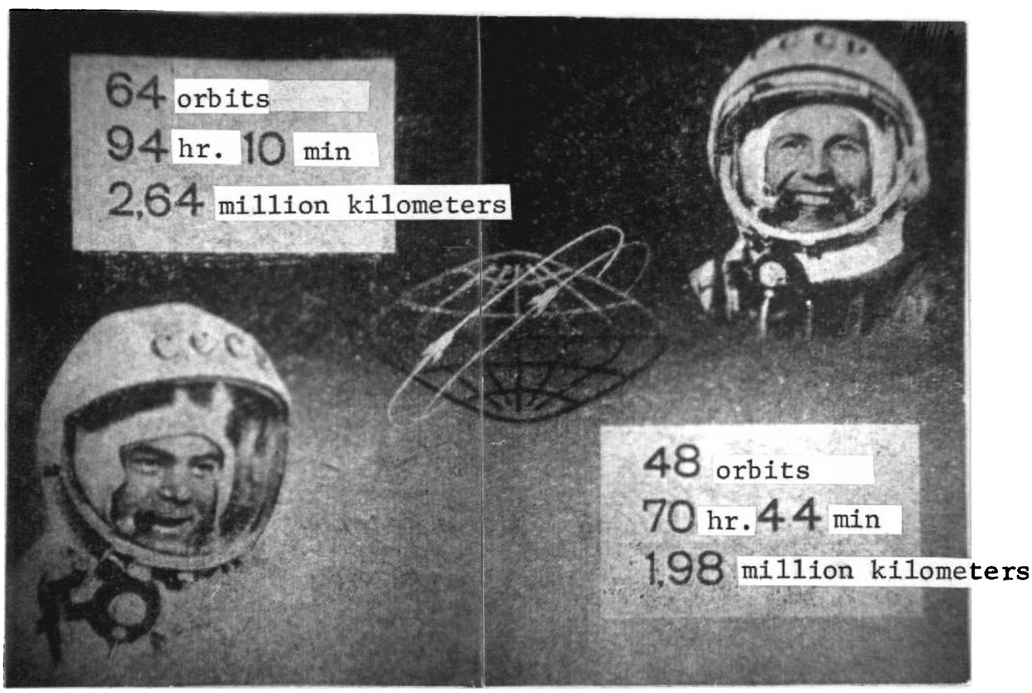


Figure 63. A. G. Nikolayev and P. R. Popovich, who completed the very first group flight in space in the Vostok 3 and Vostok 4 spacecraft.

performing this operation Popovich used the experience of Nikolaev, who was launched a day earlier and had already carried out this operation. In the free float situation the astronaut wrote, rotated around the longitudinal axis of his body and performed other motions, and did not experience any undesirable sensations. Observing the Earth's surface through the window, the astronaut was easily able to distinguish mountain ranges, rivers, and oceans. Against the cloud background, he could easily observe lightning in the form of blue flashes.

During the flight Popovich was able to maintain attitude control of the spacecraft manually without difficulty, and performed other kinds of operation, while maintaining communication with Earth and with Vostok 3. The long-duration period (up to 4 days) in space, as investigations have shown, did not produce any changes in the human body, and had no harmful effect on capabilities and coordination of motion. /154

During the flights of Nikolayev and Popovich, the first checks were made that astronauts could remain in the freely floating condition in the cabin and that they could maintain their operational activity in an unrestrained position. In addition, studies were made of the psychological functions of the astronauts, and their capabilities, and tests were made of the astronauts' life-support system. As a result of these flights, a system for life-support of group flights of spacecraft was developed.

THE FLIGHT OF BYKOVSKIY AND TERESHKOVA

On June 14, 1963, Vostok 5, piloted by V. Bykovskiy, flew around the Earth 81 times in 119 hours, covering a distance of

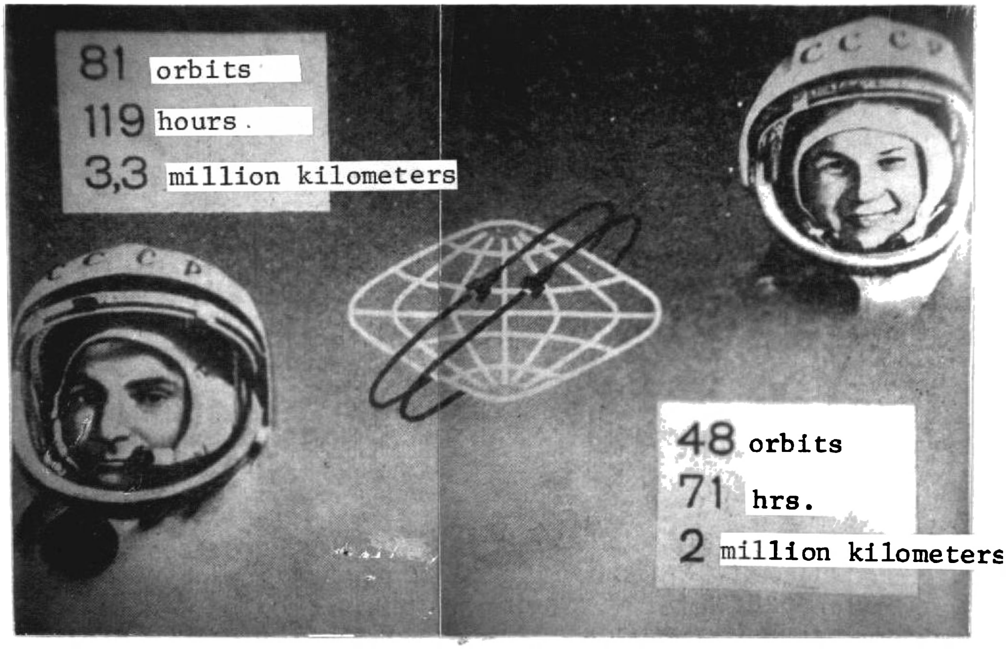


Figure 64. V. F. Bykovskiy and V. V. Tereshkova, who completed a joint flight in space in the spacecraft Vostok 5 and Vostok 6.

3 million, 300 thousand kilometers and on June 16, Vostok 6, piloted by V. Tereshkova, flew 48 times around the planet in 71 hours (Figure 64). These important flights were a major success for the effort and conceptual and mental ability of the Soviet people, the pioneers of the cosmic era, and were a tremendous scientific and engineering success; they opened up the frontiers of our knowledge of the universe and once again demonstrated the reliability and the perfection of our spacecraft. The simultaneous long-duration flight was important, and also the fact that /155
the commander of the Vostok 6 was the first woman astronaut, Valentina Vladimirovna Tereshkova, now President of the Committee of Soviet Women. One might ask the question: how does the human female cope with the complex and physically hard tasks of the astronaut?

V. Tereshkova studied rocket engineering, and spacecraft systems and instruments. She worked persistently, gave much

time to study, and to sport. Valentina Tereshkova tried hard to carry out all of the instructions of the doctors and instructors as best she could. All of this training help her to cope with the difficult work of the astronaut.

The problem of finding out about and conquering the Universe has always excited mankind. Each launch of our spacecraft was an ascent of only a single step in the space ladder, a ladder leading to other planets and other worlds. All the investigations undertaken by the Soviet space program have been directed towards increasing man's mastery over nature, and revealing her secrets. The decisive phase of this investigation is the activity of human beings, the work of people in space and on other heavenly bodies. A good deal of attention has been devoted to the development of powerful rockets capable of launching heavy objects from Earth. It has been necessary to understand and evaluate the effect of the basic factors of space flight on the human body (acceleration at launch, weightlessness in flight, and deceleration during reentry), and to check and improve the equipment giving life support to astronauts in the spacecraft. The scientists have solved the problem of controlling the spacecraft and carrying out scientific observations. To do this, the duration of flights must be increased.

After the first flights of the astronauts, it was necessary to move on to long-duration space flight. A flight of this kind, a record at the time for duration, was the flight of Vostok 5 with the astronaut V. Bykovskiy. It was already clear that in the future, it would be necessary to carry out complex maneuvers in space, to transfer from one craft to another, to engage in activity outside the spacecraft, and so on.

A new stage had arrived in the study and conquest of space. A program to accomplish this seemed huge, even in comparison with

that which had already been accomplished, There was also a very great difference between the tasks of the first artificial Earth satellite and the Vostok spacecraft in which the remarkable flight of Yu. Gagarin was accomplished.

The coming of the space age caused a readjustment of man's outlook and gave birth to new ideas. Man begins to think of new enterprises hitherto undreamed of, and to try for accomplishments which seemed nonsensical even yesterday. The scientists proposed that the first flight of man to the nearest worlds would be undertaken before the end of this century, and the knowledge of the world at large in which man lives would be one of the most interesting tasks that we would address. We should be proud that our country is playing a leading role in the study and conquest of space. /156

The flights of Vostok 5 and Vostok 6 have aroused the admiration of our friends abroad. For example, the General Secretary of the French Communist Party, Maurice Tores, in congratulating our scientists, engineers, and technicians, who have been so successful in the conquest of space, said that although this is a peaceful accomplishment, it does not mean that there are no vanquished. For example, among the vanquished are the politicians of yesterday who called Socialism a utopia and predicted the fall of a system which has just sent one of its daughters into space in the wake of one of its sons.

The press abroad reported that V. Bykovskiy had created a new world record for duration of a space flight, had made more orbits around the Earth than all the American astronauts taken together. Eugenie Cotton, the president of the International Democratic Federation of Women, said that the flight of Valentina Tereshkova was accomplished the day before the opening of the World Congress of Women and that the flight was evidence that women could be

equal with men everywhere from now on. Our nation sends its heartfelt congratulations to the crews of Vostok 5 and Vostok 6 on their successful accomplishment of a new joint space flight and our pride in their achievement which has increased the glory of the Soviet people and has enriched science. As a result of the group flights of Vostok 3, Vostok 4, Vostok 5, and Vostok 6, a broad program of scientific research has been accomplished: photography of the Sun and heavenly body, observation of the cloud cover and multiple measurements of the radiation background, allowing the position of the lower boundary of Earth's radiation belt to be accurately determined.

All these spacecraft were launched into orbit with great accuracy, and their subsequent descent to Earth was performed with even greater accuracy.

In the first space flights of the Soviet spacecraft of the Vostok series, the main emphasis was given to the study of how man would adapt to conditions of space flight. Telemetry was used to transmit to Earth the parameters describing the function of the astronaut's body. In addition, the astronaut subjectively evaluated his condition and carried out experiments to investigate the stability of his vestibular apparatus, and all his psychological and physiological capabilities.

During these flights, quite a large volume of experiments of a scientific and technical nature was carried out: data was obtained which allowed a study to be made of the physics of the Earth's atmosphere and to investigate the possibility of determining various meteorological phenomenon and forecasting their development, the solar radiation and the solar corona were investigated, problems of astronavigation were solved, and problems concerning the development of plants under weightless conditions.

FLIGHT OF THE MULTIPLE-CREW VOSKHOD SPACECRAFT

The favorable experience of space flights of the Vostok class of spacecraft made it possible to carry out expanded and complicated investigative programs on larger and more sophisticated spacecraft. In order to carry out the planned experiments, which required multi-disciplined and deeper knowledge, one person was no longer sufficient. For this reason, the Soviet Union began to plan and later to build multiple-crew spacecraft of the Voskhod class.

The first of these spacecraft, a three-man crew manned spacecraft with a crew including the Commander V. M. Komarov, a scientific specialist K. P. Feoktistov, and a doctor B. B. Yegorov, was launched into Earth satellite orbit on October 12, 1964 (Figure 65). The spacecraft differed materially from the craft previously built. The Voskhod was equipped with two braking devices. This increased the reliability for transferring the craft from a satellite orbit to a descent trajectory and allowed it to be sent to a higher orbit.

The spacecraft systems were designed to land the crew cabin with practically zero vertical velocity. The high reliability of cabin sealing allowed the crew to fly without spacesuits for the first time. Under these conditions, the three crew members were able to obtain a significantly greater volume of information in a single flight than for the flight of any of the one-man spacecraft. New data were obtained for the development of space /158
technology. Questions of group activity were investigated. Direct medical observations and instrumental psychological and physiological of investigations were made of the physiological activity of the human in flight.

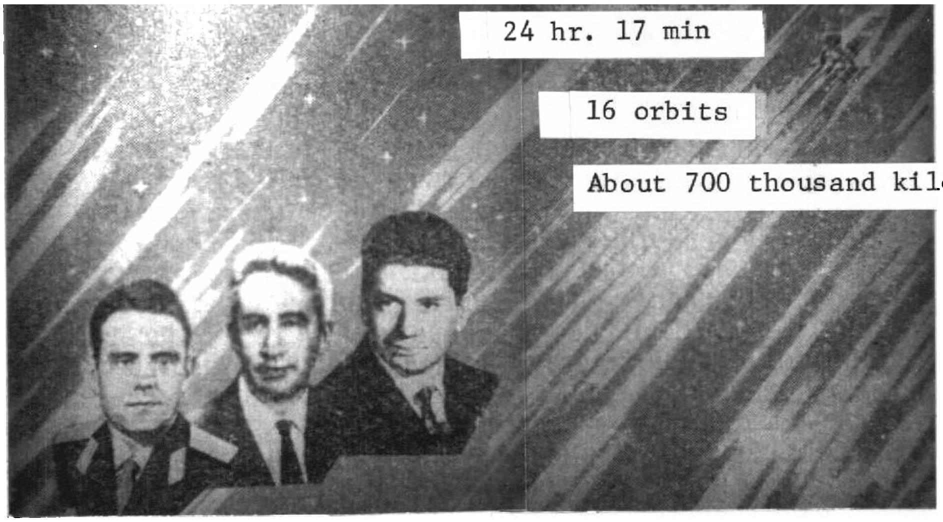


Figure 65. The crew of V. M. Komarov, K. P. Feoktistov, and B. B. Yegorov, who completed a space flight in the multi-crew Voskhod spacecraft.

Doctor Yegorov took samples in weightless conditions and then analyzed the physiological parameters of the crew members at various times in the flight.

Many scientific and technical experiments were accomplished in the realm of geophysics, and also in the field of solar physics, the structure and properties of the atmosphere and of space around the Earth. Experimental data were obtained on the structure of the limit of the horizon, necessary for choosing a reference layer in the optical range of wavelengths for the construction of navigation instruments. Observations were made on the star sky and the large stars to check the possibility of spacecraft attitude control using the stars. With this objective, astro-navigation measurements were made by means of a sextant. The crew members investigated the behavior of a liquid under weightless conditions.

Interesting observations were made by the astronaut K. P. Feoktistov, who compared his impressions of the active part of the flight from Earth and from the spacecraft cabin. After the flight he said that in the cabin everything seemed rather simpler. In addition, everything occurred almost in a routine fashion: the launch was smooth, the noise was not too great, the vibrations were not too large, the low frequencies occurred mainly in the transonic region, and the rocking motion of the rocket was slight, reminiscent of that of a train on the lines. The gravity loading, which was easily endured, naturally increased towards the end of the operation of each stage of the launch vehicle.

The flight of the Voskhod spacecraft was accomplished successfully, and the mission program was completed.

The next spacecraft, which was also based structurally on the Voskhod craft, was a two-man craft with a special transfer lock. The crew was located in a sealed cabin with outside heat shield to protect the crew and the equipment from the high temperatures during atmospheric reentry. Besides the crew couches, the cabin contained monitoring instruments for the onboard systems and control instruments for them, equipment for life-support, some of the equipment for two-way communication with ground stations, equipment to maintain heading during descent and landing, instruments for medical and biological investigations, television cameras, and supplies of food and water.

The cabin shell had three windows with heat-resistant glass and three hatches through which the astronauts could leave the ship after landing. The lock chamber, designed for the crew to exit into space and to return afterwards to the spacecraft, was located in the body of the cabin module and communicated with the cabin by means of a hatch with a sealed door. A similar hatch with a sealed door was located on the opposite side of the lock chamber.

The spacecraft contained bottles with the air supply for filling the lock, bottles with an emergency oxygen supply, and also /159
equipment for television and movie-camera purposes.

The equipment compartment contained systems for the radio equipment, for spacecraft control, for temperature control, for electrical supplies, and also a liquid retro-motor unit and a spare solid-fuel retro-motor. On the outside of the spacecraft shell, there were the motors for attitude control, and bottles of compressed air and oxygen to be used for ventilation of the astronauts' spacesuits.

The spacecraft design called for separation of the transfer chamber from the equipment compartment after the spacecraft entered its Earth descent trajectory.

EXTRA-VEHICULAR EXCURSION OF A HUMAN IN SPACE

On March 18, 1965, a powerful launch vehicle injected the Voskhod 2 spacecraft, piloted by a crew consisting of the commander P. I. Belyaev and the second pilot A. A. Leonov, into orbit from Earth orbit. On the second revolution Leonov made an excursion in a special suit into free space (Figure 66), withdrew from the spacecraft to a distance of more than 5 m, and, having completed observations called for in the work program, returned to the cabin.

An excursion into outer space was an historic new step in the conquest of space. Of this important experiment, the astronaut Leonov has said that, immediately after being placed in orbit, they began to prepare for the experiment. Before entering the lock chamber, while still in the spacecraft cabin, he was assisted by the commander in donning the backpack with self-contained life-support systems.

Figure 66. P. I. Belyayev and A. A. Leonov, who flew in the Voskhod 2 spacecraft (the first excursion of a human from a spacecraft into space was made by Leonov).

They checked the operation of the hardware, systems, and recording equipment. They equalized the pressure in the lock and the cabin. Then they opened the hatch from the cabin to the transfer chamber and Leonov passed through into the chamber; then the commander closed the spacecraft hatch. After reducing the lock pressure Leonov opened the outside hatch. A bright shaft of sunlight filled the transfer chamber. Finally everything was ready, and he was able to go out into outside space.

Leonov put his head outside the exit hatch. The immensity of space appeared in front of him in all its beauty. The Earth swam majestically in front of his eyes. Without hurrying, he moved out through the hatch, and then, pushing off gently from the hatch, he moved away from the spacecraft. The halyard by which the astronaut was attached to the spacecraft extended to its full length, and his motion away from the craft was arrested.
As Leonov narrates it, he expected to see sharp contrasts of /160
light and shadow, but there was nothing of that kind. The parts of the spacecraft in the shadow were quite well illuminated by sunlight reflected from the Earth. Leonov tugged the halyard a little towards him and slowly began to approach the spacecraft. Later on he said that by becoming adjusted, he could move rather precisely and with coordination in these unusual conditions.

This experiment showed the whole world that a human being is able not only to accomplish flight, located within a spacecraft, but can operate actively in space, and can work in space.

In the flight the astronauts worked in spacesuits. These spacesuits were qualitatively new. In them a person could move outside the spacecraft into deep vacuum, with both a deep degree of frost and strong heating. The suit for an excursion into space was equipped with a self-contained life-support system.

The system maintained a given pressure and normal temperature inside the suit, eliminated the surplus moisture, and supplied the astronaut with oxygen for breathing. The astronaut's visor was equipped with a special light filter to protect his eyes from the bright rays of the Sun. Inside the helmet were the microphones for communication with the commander, and in case of need, the astronaut could converse directly with Earth.

Leonov's excursion into space was watched by people on Earth, viewing their television screens. A camera, mounted on the outside of the lock compartment, showed the astronaut operating during the entire time of his period outside the spacecraft.

The engineers were not clear how a person would behave in conditions of unconfined space and whether he would move away from the spacecraft at all. They thought that immediately when he emerged from the spacecraft, his gloves might be "welded" to the spacecraft skin. How would the astronaut move himself outside the spacecraft? Would he be able to keep his orientation in space? Would the halyard hold him if he departed abruptly from the spacecraft? All these and other questions had put Leonov roughly in the same psychological situation in which Gagarin found himself during the first flight. However, Leonov's situation was somewhat eased, since he was not alone. Belyaev remained inside the spacecraft and maintained a continuous two-way communication. He directed the activities of Leonov and, if need be, was ready to come to his aid. The total time spent by Leonov in space was twenty minutes, including twelve minutes outside the spacecraft.

During the process of the excusion into open space, many new structural features were checked: the lock method, the reliability of operation of the control systems, the effectiveness of the self-contained life support system outside the

spacecraft; the technique for control of the astronaut's motion during the excursion was tested, his ability to work in conditions of free fall, and to return to the spacecraft. When the whole program was accomplished, Belyaev was given permission to land. Voskhod-2 had a duplex redundant landing control. In case of malfunction of the automatic orientation system, including the retromotor, a landing could be performed using a manual control system. The commander of the two-man spacecraft tested this system in operation. After the required calculations, Belyaev oriented the spacecraft manually and fired the retroengine unit.

The need arose to transfer to manual control in connection with failure of the automatic solar orientation system. In these conditions, the crew kept its coolness and self-restraint, as well as their high professional competence as pilots and capabilities after a long period in space. The first test using manual control demonstrated the crew's splendid training, their ability to adjust to a new situation rpaidly and correctly, and to take control of the spacecraft systems.

This flight opened up a new page in this history of the conquest of space. It confirmed the operational reliability of the spacecraft lock system and the convenience in use of the lock chamber, the reliability of the spacesuit under conditions of total vacuum and the error-free operation of the communication between the astronaut and the spacecraft in conditions where he was floating freely in space. Much attention was given in the flight to the development of team work on the part of the crew, which consisted of two people, during a long flight and in solving the problem of conducting an excursion into space. The crew of the Voskhod-2 coped brilliantly with the tasks assigned /162
to it. Valuable data were obtained which were subsequently used by the American and Soviet astronauts in performing excursions into space and when transferring from one spacecraft to another.

FLIGHTS IN THE SOYUZ 1, SOYUZ 2, AND SOYUZ 3 SPACECRAFT

Continuing with their planned conquest of space, the Soviet engineers and scientists created a new type of spacecraft. This was the Soyuz series, which, besides allowing complex investigations of near-Earth space, also opened up a new phase in the development of astronautics — that of setting up and using orbital laboratories. Each experiment has verified a logical step in the development of Soviet astronautics. Satellites of the Cosmos series were used to test automatic docking in orbit. This opened up possibilities for assembly of large spacecraft and scientific space laboratories in Earth orbit, where the separate parts would be placed in orbit by small launch vehicles, and later assembled without the direct intervention of humans.

On April 23, 1967, a new spacecraft, Soyuz 1, was placed in Earth orbit. Its pilot was the USSR astronaut, Hero of the Soviet Union, Engineer Colonel Vladimir Mikhaylovich Komarov (Figure 67), who had previously made a flight in the Voskhod spacecraft. The objective of the Soyuz 1 flight was to test a new manned spacecraft and to develop systems and features to adapt it to spaceflight conditions.

During the flight, Komarov conducted a full program of tests for the new spacecraft, and made many valuable observations on the layout of the spacecraft systems. This has contributed significantly to the successful flight program of spacecraft of the Soyuz series. After performing the flight, Vladimir Komarov was killed during the landing, owing to malfunction of the parachute system. He was made a Hero of the Soviet Union a second time, posthumously.

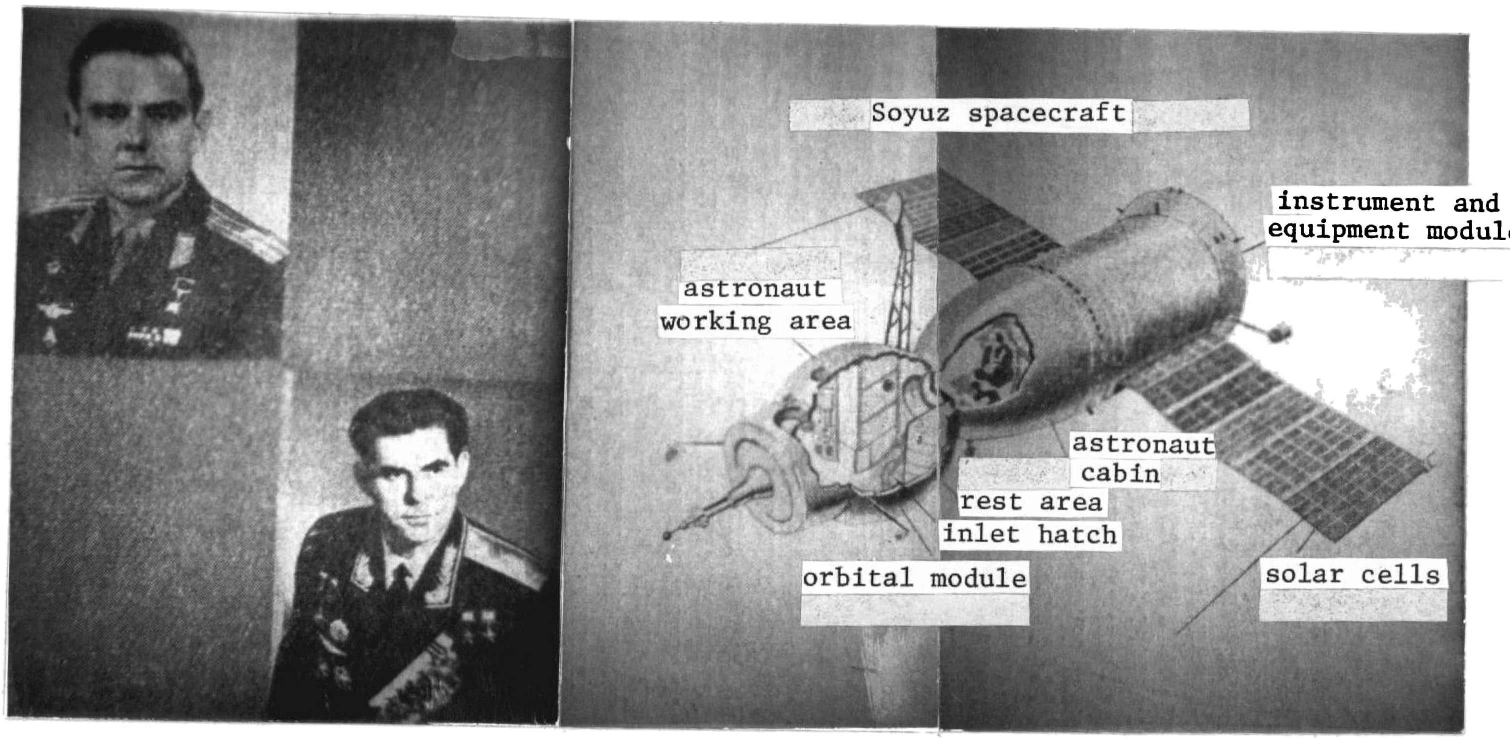


Figure 67. V. M. Komarov and G. T. Beregovoy, who flew in the Soyuz spacecraft.

After improvement of the Soyuz spacecraft, it was necessary to test the spacecraft systems under actual flight conditions. This task was assigned to the test pilot G. T. Beregovoi (Figure 67), a Hero of the Soviet Union. On October 26 1968, the Soyuz 3 spacecraft was launched, and one day later, the unmanned Soyuz 2 was placed in orbit. During the four-day flight of Soyuz 3, a maneuver was performed several times which used the automatic and manual guidance systems: a rendezvous with the unmanned Soyuz 2 was performed twice, a full set of spacecraft system tests was accomplished, as well as a large volume of scientific investigations and observations, including: observation of the star sky, the Earth, and the stars, photography of the cloud cover of the Earth's daytime and twilight horizons, observation /164
of typhoons and cyclones, as well as medical and biological investigations under spaceflight conditions.

The Soyuz 2 and Soyuz 3 flights confirmed the validity of the technical design on which the new type of spacecraft, capable of a wide range of manuevers, was based. Valuable new data were obtained from the scientific and technical experiments.

Sometimes the question is asked: is it valid for a human to conduct space investigations onboard a spacecraft, and could these results be obtained by automatic methods? For all of the achievements of contemporary automation and telemetry, the human brain remains the most reliable programmable device, for it is the only one than can rapidly analyze a new set of circumstances and come up with a solution during the experiment. The human part in accomplishing scientific programs onboard manned spacecraft addresses important matters of preference, such as conscientious choice of the objects to be investigated. However great the capabilities of satellites and automatic spacecraft may be, they cannot match the creative capability of the human. The robot cannot investigate what is, in principle, unknown; he

is capable of studying only matters which are already known to man in some degree. And only the human can analyze the results obtained in the course of the investigations, perceive correct solutions to unforeseen sets of circumstances, and fully utilize the possibilities that present themselves for studying the surrounding environment.

Manned spacecraft, in combination with automatic equipment, can carry out new space investigations, take continuous regular scientific information and data, set up complex scientific, engineering, medical and biological experiments, and also equip expeditions more reliably, not only for Earth orbit, but also for remote space journeys. This is, in fact, the Soviet program of space investigation.

Structurally, the Soyuz spacecraft consists of the astronaut's cabin (called the launch module), the orbital module, and the instrument and equipment module. While in the launch module, the astronauts are launched, land, and carry out the required maneuvers in orbit during docking of the spacecraft. The cabin of the Soyuz spacecraft has a number of improvements in comparison with those of earlier spacecraft. Its shape, a segment of a conical body, is reminiscent of an automobile headlight, and provides aerodynamic lift during atmospheric flight. A method of varying this lift is provided to control the flight during motion in the atmosphere. The descent trajectory uses aerodynamic force, as has been mentioned, thus reducing the loading on the crew during the descent, to below about 3 — 4 g (in /165
comparison with 8 — 10 g in ballistic entry of previous spacecraft).

The orbiting module is the scientific laboratory in which the astronaut conducts the scientific observations, sleeps, performs the required set of physical exercises, and takes his food. By using the orbital module as a lock chamber, the astronaut can make an excursion into space.

The docking module is mounted at the forward end of the orbital module.

The function of the instrument and storage module is to store onboard equipment and spacecraft engines needed for orbital flight. The module contains the units of the thermal control system, the electrical supply system, equipment for remote communication and radiotelemetry, and instruments of the attitude control system and of the computer system for control of spacecraft motion.

THE ORBITING LABORATORY

A major step in the conquest of space was the group flight of the spacecraft Soyuz 4 and Soyuz 5, launched on January 14 and 15, 1969. This was the first simultaneous spaceflight of four astronauts: V. Shatalov, B. Volynov, A. Eliseyev, and E. Khrunov. On January 16, millions of people on our planet became witnesses to this new experiment, which laid the foundations for the creation of an orbiting laboratory.

The Soyuz 4 spacecraft was launched into orbit with the commander, Shatalov. Next, Soyuz 5 was launched, carrying three astronauts. The spacecraft carried out rendezvous and docking. Thus, assembly of an experimental orbiting laboratory (Figure 68) was carried out in space for the first time.

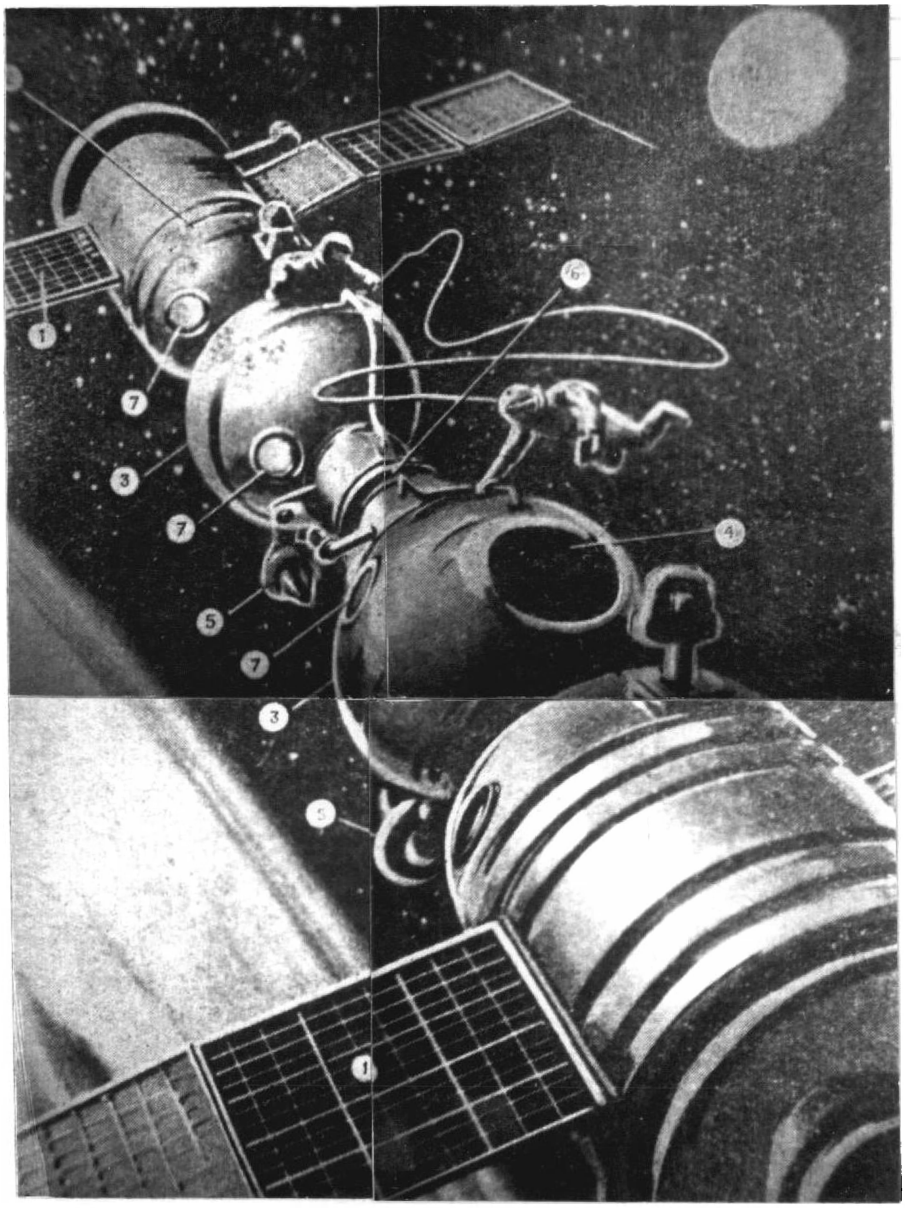


Figure 68. The first space laboratory.

1- solar cell panels; 2- the descent capsule; 3- the orbital module; 4- the lock for excursion into space; 5- antenna; 6- location of the docking joint; 7- illuminating lamps.

Ordinarily, during operations for maneuvering and rendezvous of two space objects, one of them is the active partner, and the other passive. For example, during the flight of the Soyuz 4 and Soyuz 5 spacecraft, the rendezvous maneuver was made by Soyuz 4. The ignition of the launch vehicle carrying Soyuz 5

was executed at an accurately computed time. The engines and the automatic control system injected the spacecraft into an orbit close to that of Soyuz 4. On command of the programmed timing device, the antennas of the onboard radio equipment and solar panels were deployed, and the attitude control and flight control systems were switched on.

The period of joint flight of the two Soyuz spacecraft began. The most crucial stage of the flight took place in the next few days. The onboard radar system for search and homing was switched on, and accomplished the more distant section of spacecraft rendezvous. Starting at a distance of 100 m, the spacecraft commander Shatalov took over control of the spacecraft motion. He switched on and off the motor jets controlling the linear velocity of the spacecraft, i.e., he decelerated or accelerated it, and cancelled the lateral velocity. At the moment of contact between the docking modules of the spacecraft, the relative velocity had to be equalized. The relative velocity before contact must not exceed 0.25 m/sec.

The experimental spacecraft had four compartments in which the astronauts could live and work in comfortable conditions. It was equipped with all the necessary scientific apparatus, and had internal telephone communication and radio television equipment for external communication with Earth. The total volume of the working space in the two spacecraft was 18 m^3, and the spacecraft systems operated as a single entity after docking.

On January 16, Khrunov and Eliseyev, the crew members of Soyuz 5, donned their spacesuits (Figure 69) and went out into space through the lock of the orbital module. They spent about an hour in space. Then they returned into the orbital module of the Soyuz 4, doffed their spacesuits, and occupied their new working space. This was the first time that transfer from one

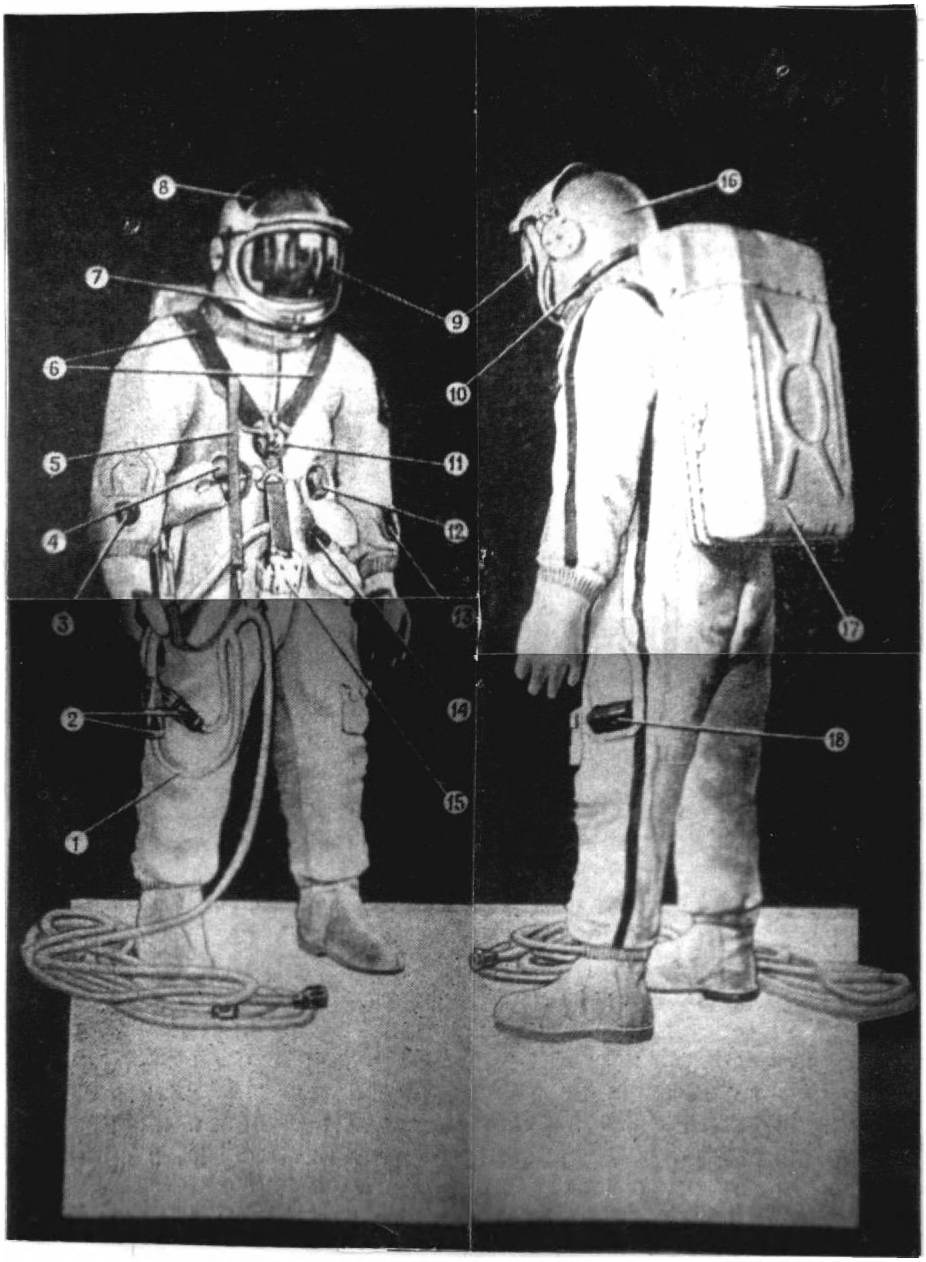


Figure 69. An astronaut's spacesuit.

1- telemetry halyard; 2- joints; 3- manometer; 4- communication joint; 5- back-up halyard; 6- haversack suspension system; 7- bracket for visor; 8- light filter; 9- spacesuit window visor; 10- jointing ring; 11- lock for haversack suspension system; 12- valve for selecting spacesuit operating mode; 13- mirror; 14- valve for maintaining required pressure; 15- control box for life support system; 16- sealed helmet; 17- life support system backpack; 18- manual valve for switching on emergency oxygen supply.

spacecraft to another had been performed in artificial Earth orbit. Thereafter, the spacecraft were undocked and continued their further missions separately.

Astronauts Khrunov and Eliseyev have reported that during the transfer they met no major difficulty, and performed the manuever confidently. After the flight program was completed, Soyuz 4 made a landing with three astronauts, and was followed by Soyuz 5. During the flight, both the men and the automatic systems took part in creating an experimental space laboratory and making it function. In the future, naturally, there will be artificial self-adaptive systems which will perceive changes in the system and its changing parameters, and the intervention of humans will be required only in exceptional cases. But no kind of automatic system can fully replace the human when decisions have to be made after information is gathered, particularly in non-programmed situations. It is true that a machine has quite important advantages over a human: it does not suffer from fatigue, annoyance, unreliability, fear, and other psychological phenomena. However, one should not forget the qualities that are exclusively human: will-power, a creative mind, and a high moral spirit. The machine can never displace a human in the sphere of creative activity and, therefore, in making up space systems, it is a question of optimizing the most appropriate combination of the properties and qualities of the human and the machine. Such a combination was demonstrated during the flight of the Soyuz 4 and Soyuz 5 spacecraft.

THE STELLAR SQUADRON /169

Later on, the Soviet Union carried out a group flight of three spacecraft: Soyuz 6, Soyuz 7, and Soyuz 8 (Figure 70).

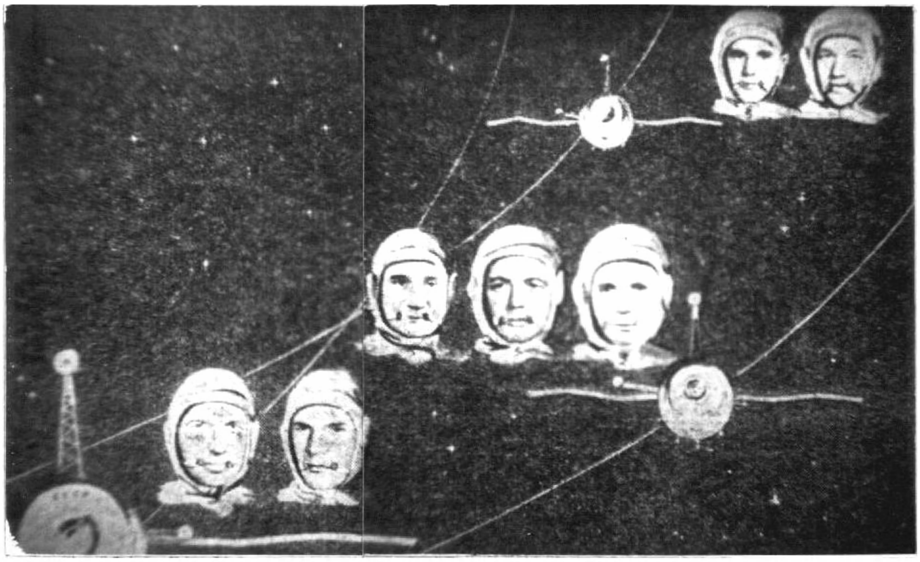


Figure 70. Crew of the stellar squadron, who completed flights in the spacecraft Soyuz 6, Soyuz 7, and Soyuz 8, consisting of G. S. Shonin and V. N. Kubasov; A. V. Filipchenko, V. N. Volkov, and V. V. Gorbatko; and V. A. Shatalov and A. S. Eliseyev.

On October 11, 1969, Soyuz 6 was placed in Earth orbit with a crew of G. S. Shonin and V. N. Kubasov; on October 12, Soyuz 7, with a crew of A. V. Filipchenko, V. N. Volkov, and V. V. Gorbatko, was launched; and finally, on October 13, the third spacecraft, Soyuz 8, was launched with its crew, V. A. Shatalov and A. S. Eliseyev (for them, it was the third space flight).

This was the first spaceflight of three manned spacecraft in Earth orbit.

In these spacecraft, the pilots interacted with a sophisticated complex of automatic devices, including different control systems, for obtaining and operationally processing information and communication.

During the flight, tests and checks were made of the spacecraft systems, manual systems for guidance, attitude control and stabilization were operated in orbit, and checks were made of self-contained navigation systems. Joint maneuvers of the spacecraft were conducted, and the experience deriving from this new experiment served to build a manned space complex in Earth orbit.

During the several days of group flight of the three Soyuz spacecraft, qualitative solutions were found for new problems associated with the creation of manned orbiting space systems and the development of interaction of spacecraft when engaged in a wide range of maneuvers in Earth orbit.

The fact that Soyuz had a number of modules (the astronaut's cabins, the orbital module, designed for scientific investigation and sleeping) made it possible to accomplish a varied program of scientific investigation. One important assignment accomplished on Soyuz 6 was tests to investigate different methods of welding metals under conditions of deep vacuum and weightlessness. Thin sheets of structural materials were welded, stainless steel and titanium, and cutting was done of stainless steel, titanium, and aluminum, as well as processing of nonmetallic materials; the behavior of a drop of liquid metal and of a welding bath was investigated under conditions of weightlessness.

As has been mentioned, the development of astronautics poses problems for the designers and scientists, associated with building large orbiting laboratories and interplanetary spacecraft from parts injected into Earth orbit. The accomplishment of such projects entails a requirement to join metals in different ways, including welding. Therefore, in developing an experimental welding facility which would allow the possibility of performing welding work in space to be checked, the scientists started from

the premise that the welding methods to be tested would be able to join the various metals and alloys used in the construction of spacecraft. The "Vulcan" self-contained welding unit used on Soyuz 6 consisted of a welding unit consisting of the operator's welding handle and a turntable with specimens of metals to be welded, an instrument unit with an electric supply, a protective case, and a remote control panel. The unit provided welding by means of a compressed arc (low temperature plasma) electron beam, and by means of floating electrodes.

Before beginning welding in space, the spacecraft commander closed the lock port into the cabin; then he went through the operation of sealing the orbiting module. Then the engineer switched on the welding equipment and performed all three types of weld in succession.

On completion of the experiment, the orbiting module was again sealed, the pressure was equalized between the descent capsule and the orbiting module, the lock manhole was opened, and the crew members came out into the orbiting section to inspect the welding facility and the weld specimens.

The experiments conducted on this flight showed that it is possible in principle to weld metals under conditions of weightlessness and deep space vacuum.

During the flight of the Star Squadron, a broad program of scientific investigations was also conducted, including specific ways of using manned orbiting systems for production purposes. Experiments were done to study typical geological sections of the Earth with the objective of determining whether it is possible, in principle, to identify regions with deposits of raw materials, to determine the reflective properties of forest areas, desert areas, and other sections of the Earth's surface.

Boundaries were determined for distribution of snow and ice cover. During the flight, the astronauts took a considerable number of photographs and movie photographs of the Earth's continental areas, oceans, and cloud cover, performed astrophysical observations and experiments (in particular, they determined the polarization of solar rays reflected by the atmosphere), measured the illumination due to the Sun, conducted experiments to determine the true brightness of stars, etc. The program of medical and biological investigations was continued with the study of peculiarities in the physiological processes in the human body under conditions of weightlessness in spaceflight. A study was made of gas and energy volume and energy rates, as well as the functional state of human respiration and blood circulation after accomplishing different kinds of work.

During the group flight of Soyuz, a large number of complicated maneuvers was accomplished, many of them being performed by the astronauts manually. The total number of such maneuvers was more than thirty. This was considerably more than in previous flights. The maneuvers were accomplished in two ways. In the first case, the control center, knowing the location of a spacecraft in orbit, communicated to the crew in their transmissions how the operation should be conducted, in order for the spacecraft to draw close together. The astronauts performed these operations according to the instruments, in part even without seeing the spacecraft. In the second method, the control center did not interfere in the action of the astronauts. They themselves decided how to move and how to accomplish the manuevers. Only self-contained means were used and radio conversation between the astronauts. As a result, Soyuz 6 and Soyuz 8 approached close to the base spacecraft, Soyuz 7.

The astronauts' life support systems maintained comfortable conditions in the spacecraft living compartments. During the whole flight, constant medical supervision was maintained over the astronauts' state of health.

In the flights of astronauts Popovich and Nikolaev, the possibility of operating two manned spacecraft was verified, with mutual communication between them, and they were injected accurately into close orbits. In the flight of Beregovoy, the mutual maneuvering of the manned spacecraft and the unmanned spacecraft was accomplished without docking in orbit, and now in the flight of Soyuz 4 and Soyuz 5, a limited maneuver was accomplished in space with manual docking, thus forming the first orbiting laboratory. In this way, from experiment to experiment, the astronauts carried out increasingly complex operations.

The flight of the Soyuz 6, Soyuz 7, and Soyuz 8 spacecraft revealed new possibilities in spacecraft control and provided valuable observational material concerning space. The spacecraft were equipped with sextants, electronic computers, and other devices which could determine the position of the spacecraft and its motion in space.

An important task of the flight program was also to develop methods of interaction of the group of spacecraft with ground command and measuring points, located in different regions of the Soviet Union and in the scientific research centers of the Academy of Sciences of the USSR, which are located at a number of points in the Pacific Ocean. The Molniya communication satellites were included in the system for transmitting commands and measurement information. The results of the joint flight of the Soyuz spacecraft showed that the control arrangement adopted was highly efficient.

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After completing the full volume of their scientific and engineering investigations and experiments, the Soyuz 6, Soyuz 7, and Soyuz 8 spacecraft landed in the assigned region of the Soviet Union on October 16, 17, and 18, respectively.

On the 80th orbit of the Soyuz 6, all the required operations for evacuating the crew were completed, and the crew took their places in the descent capsule. The landing was carried out using manual control. The retroengine was fired, the spacecraft orbital speed diminished, and the vehicle transferred to a flight trajectory towards the Earth. Then the orbital and equipment storage modules separated from the descent capsule and the landing control system was switched on. The system first rotated the capsule in pitch to accomplish atmospheric entry at a specific angle of attack and provided the required aerodynamic features by rotating the spacecraft about its longitudinal axis, this being accomplished by special motors. At an altitude of about 10 km, the lid of the parachute container was blown off, deploying the braking parachute and the main parachute system. After switching on the soft landing motor engine, the spacecraft came gently to Earth.

The descent of Soyuz 7 and Soyuz 8 took place in a similar manner. Control of each spacecraft was accomplished by means of engines of four different kinds. A correction engine (first type) was used to accomplish maneuvers in orbit. It also served as a retroengine for the descent. Low thrust engines (second type) were used to rotate the spacecraft about its center of mass to orient it in a given direction.

The motors of the third kind were also designed to rotate the spacecraft about its mass center. They were used also for small translational adjustments of the spacecraft, required for /173
mutual maneuvering of spacecraft at a small distance apart.

The engines of the descent control system (fourth type), located directly on the descent capsule, were used for programmed rotation of the spacecraft before atmospheric entry, and also for roll control for stabilization relative to all the axes during the atmospheric flight.

These successful experiments will undoubtedly find rapid practical application.

In this flight, the beginnings of space metallurgy were laid. The remarkable flights of the Soviet spacecraft have been highly praised by the party leadership and the administration, and by all the Soviet nation.

There have been many responses in the Soviet Union and abroad to the flight of this squadron, and the first Soviet space "electric welding shop" has attracted a great deal of attention.

Scientific reviewers in the press abroad have commented that the complex program of the group flight included important scientific and engineering experiments - in particular, space welding - and that the experiments accomplished during the flight are a major accomplishment. Records were established concerning the number of astronauts and spacecraft injected into orbit, and a record was made with respect to sequential launch of spacecraft from one site and for the radio and television communication between Earth and three spacecraft.

The American newspapers have written that this new success for Soviet science and technology is a remarkable achievement in space which not only has great technical significance, but also holds out great promise for future flights. They also

mentioned that the USA will be ready only in 1972 to test three types of metal welding in space, and this experiment was verified during the flight of Soyuz 6.

LONG DURATION FLIGHT

On June 1, 1970, at 22 hours Moscow time, a launch vehicle lifted off with the Soyuz 9 spacecraft (Figure 71). The spacecraft was piloted by a crew composed of the commander, Hero of the Soviet Union, pilot astronaut of the USSR, A. G. Nikolayev and the flight engineer, Candidate of Technical Science, V. I. Sevast'yanov.

The program for this flight is distinguished by the large number of dynamic operations associated with spacecraft attitude control. Several tens of such operations were performed. The conduct of many experiments and observations required preliminary orientation of the spacecraft followed by stabilization of its position by means of gyroscopes. As a rule, as Nikolayev has mentioned, the operations were accomplished manually, and sometimes, with transfer to automatic control.

The spacecraft was rotated almost daily in order to orient the solar cell panels towards the Sun. To do this, the spacecraft was first oriented in the appropriate way, and then stabilized in this position by creating rotation relative to the orientation axis. All the rotation was provided by manual control. During the flight, the spacecraft was oriented several times towards the Earth, above its shadow side. Here automatic, semi-automatic, and manual control methods were used. During the flight, tests were made of a number of instruments used in the attitude and spacecraft motion control systems. In particular, a check of spacecraft roll was done when it was

Figure 71. A. G. Nikolayev and V. I. Sevast'yanov, who made flights into space in the Soyuz 9 spacecraft.

pointed towards the Sun at various assigned angles, using wide angle optical indicators. This control, according to Nikolayev, was very effective, and subsequently, a technique was developed for optimum control of a spacecraft using this indicator. The method was used in performing the so-called coning rotation. These rotations were accomplished with the space- /175 craft at a certain angle towards the Sun in order to provide the required thermal conditions for the spacecraft and to maintain normal operating conditions for the buffer cell during a period of relatively low onboard electrical energy requirement.

During the flight, orbital correction was carried out three times. Here both manual and automatic methods were used. The experiments were continued to develop methods and means for self-contained navigation. The period of rotation was determined by measuring the flight altitude, and angular measurements were made on ground fixes and stars. From the results of these measurements, all the spacecraft orbital elements were determined and the required trajectory corrections were calculated. Optical methods were used to determine accurate characteristics of the gyroscopic instruments of the attitude control and stabilization system. An investigation was conducted into the nature, dynamics, and brightness characteristics of luminous particles and a test was

done of lamps to provide reliable operation of the optical sensing elements during the whole flight.

During the flight, an estimate was made of the effect of aerodynamic and gravitational perturbing moments on the dynamic characteristics and control of the spacecraft. Experiments were performed to study the structural characteristics of the spacecraft, and, in particular, the strain due to vacuum conditions and the one-sided heating from the Sun was measured.

On the 188th orbit, when the spacecraft was located above the Indian Ocean, a complex experiment was conducted in which the Meteor weather satellite and the scientific research station Akademik Shirshov took part, besides the Soyuz 9 spacecraft. The experiment made it possible to investigate atmospheric formations in this region of the Earth's surface.

During the flight, the following were performed repeatedly: observation and photography of clouds in order to study their three-dimensional structure, the determination of boundaries for snow deposits, observation of gales, storms, and tropical cyclones, photography of geographic and geological items on the Earth's surface within the territorial USSR, the objective being to improve existing and create new geological charts to assist in searching for useful minerals, daily observation of the Earth daytime and twilight horizon, and also of the Moon in the background of the Earth's horizon. The data from these experiments enabled altitude profiles of the brightness of the horizon and the atmosphere to be calculated, and these, in particular, made it possible to increase the accuracy of the onboard navigational measurements. The spectral brightness of various objects in the visible part of the spectrum was investigated in order to create new systems for attitude control and to determine the upper

boundary of clouds, the objective being to develop methods for processing weather information taken from weather satellites.

During this flight, as was done in previous flights, observations were made on luminous particles through the spacecraft window, and their size, brightness, and speed of movement were determined.

The astronauts also performed several biological experiments to study the effect of weightlessness on growth, development, and heredity for various types of living organisms. During the flight, valuable data were obtained on tests of different systems for life support of the crew.

The Soyuz 9 spacecraft was basically analogous to previous craft of the Soyuz type, in regards to its structural characteristics and its layout. However, because of the tasks of a long duration flight, several measurements were included to improve the astronauts' working and sleeping conditions. The spacecraft also consisted of three main compartments: the orbital, descent, and equipment storage module. It lacked the apparatus and equipment to accomplish rendezvous and docking with other spacecraft. The orbital module on Soyuz 9 functioned as a laboratory for scientific experiments and investigations, and also as an area in which the astronauts could relax, sleep, and take their meals. It housed the main scientific apparatus and facilities. It contained a system for recording the thermal control, which maintained the climatic conditions at Earth values during the long flight. To maintain the normal physical condition and high efficiency during a long flight in the weightless state, the astronauts performed a set of exercises in special costumes in a special gymnastic area. The area had two shock absorbers which could produce a loading on the human body. The orbiting

module of the spacecraft was a prototype of the scientific laboratory and living areas of orbiting laboratories. The work of the astronauts in this module would help to clarify specific features and choice of logical methods for the conduct of investigations in space conditions.

The spacecraft descent capsule had two variants. The long flight and the large volume of scientific investigations required a partial change in its layout. The third couch was replaced by scientific apparatus, cassettes for movie and still photographs, and a magnetic tape store for recording results of onboard experiments in orbit. The informational records and a number of scientific instruments for the biological experiments had to be returned to Earth. Just as in the previous spacecraft, the descent capsule had a panel with instrument meters and signal lamps, keys and knobs for controlling the spacecraft and its systems. The descent from orbit was designed to incur low loading (3 g at most). This was important for astronauts who had lived for a long period in the weightless state. In addition, the descent capsule had shock absorbing couches and retrojet motors to reduce the loading at the moment of touchdown following the parachute descent.

The instrument and storage module of the spacecraft was similar to the corresponding modules in previous spacecraft. The module contained apparatus and equipment dealing with the orbital flight and the descent from orbit after accomplishing the flight program, and including a correction motor, attitude control motors, the thermal control system, solar cells, and the antennas of the radio and telemetry systems.

During their 18-day flight, the crew of Soyuz 9 checked the strength and reliability of technical features and the endurance of the human being. A careful study was made of human behavior under conditions of a long period in the weightless state and when restored to the Earth's gravitational field.

It is correct to say that, with the contemporary state of the level of development of technology, it is unattractive to launch expensive space systems for several days. They must remain there for a long time. Orbiting laboratories are required and transport spacecraft will be used to equip them with all that is necessary and to exchange the crews. In addition, in the not too distant future, spacecraft will be sent to other planets; this is unavoidable. Therefore, an increase in the duration of spaceflight is a logical result of the general progress of astronautics. However, the possibility of further manned flights is determined not only by the improvement of space technology, control and navigation systems, but also by the level of development of sciences such as space biology and medicine, which seek to provide conditions suitable for living and working on manned spacecraft. The effect of various spaceflight factors on the human body must be studied, and ways and means of protecting humans from undesirable effects must be developed. A long stay in the weightless state, in the opinion of several specialists in the field, may be a barrier to long duration spaceflight. In the history of aviation development, there have been many barriers (e.g., the supersonic barrier and the thermal barrier). To overcome these required basic scientific investigation and special experiments. In astronautics, the problem of launch into space and Earth re-entry has required solution of complicated problems in engineering psychology, associated with spacecraft control. The creation of the optimum "man-machine" system should foresee all that is new and unknown and can be met

in space. Therefore, 424 hours in the weightless state was not only a record, established by the crew of Nikolayev and Sevast'yanov in the flight around our planet, but was also a major scientific experiment. It generated valuable information which must be studied at length.

In addition to a study of the long duration effect of space flight factors on the human body, an important task was to investigate the process of human transition, after a long stay in the weightless state, to conditions of Earth gravity. The hazards /178 with regard to adaptation to Earth conditions proved to be invalid, although the adaptation difficulties were not all anticipated. After eight days of weightlessness, the entire body suddenly became heavy, according to the astronauts' account. The sensation was like being on a centrifuge under the action of a small loading.

The flight had its share of funny incidents. The astronauts slept in special sleeping bags. Sometimes one of them would float out of his sleeping bag in his sleep, and when he awoke, he would find himself on the ceiling of the orbiting module.

Many specialists had been waiting impatiently for this flight: biologists, medical men, and structural engineers. The point is that, prior to the flight, it was an open question as to whether a human could live and operate for a lengthy period in space conditions, in conditions of weightlessness, since the first flights of the Soviet and American astronauts had shown that certain changes in the human body appear in the weightless state, first in the blood, muscular and cardiovascular systems. These changes, called functional adjustments, are not an illness, and vanish completely with time. However, they can reduce the stability of the human with respect to the loading arising during the landing. Therefore, it was necessary to prepare and carry

out both ground experiments and orbital experiments. The entire development of mankind has been associated with residence in conditions of Earth gravity. A human exposed for a long period to a weightless state quite unusual for him takes the risk of "outgrowing" Earth conditions, i.e., a powerful muscular system will not be required, and he begins to grow weak and to degenerate. The same thing may occur also with the human blood system, since calcium is gradually eliminated from the body. To prevent the human from "outgrowing" Earth conditions, the specialists plan to create an artificial gravity force on the spacecraft, i.e., a rotating system in which the person will live. The centrifugal force arising during rotation will simulate a constant load, that of Earth gravity. However, because of the addition of the rotary and orbital motion, the spacecraft will change in its orbit, and this must be corrected. Besides, such a system will have a large mass. Therefore, the main effort is directed towards combating the phenomenon in the weightless condition, and mainly goes into development of methods of physical training. In addition, it is known that a medium deficient in oxygen, typical of high altitudes, allows the retention of calcium in the blood tissue to be stabilized and increases the volume of the blood circulation, i.e., it counteracts the phenomena which accompany a long period in the weightless state.

The Soviet people and the world community have followed with great interest the orbital flight of the Soviet Soyuz 9 spacecraft, results of which were published in detail under captions such as: "American record broken"; "Soyuz 9 has begun its third week in space"; "We greatly envy the Russians their Soyuz"; "Experiments like these are necessary to design and build orbiting space laboratories"; "All records are broken, Soyuz 9 has returned to Earth"; "Great contribution to science". This was the kind of comment in the press abroad on the occasion /179

of the Soyuz 9 flight. The Russian people solemnly honored the astronaut heros and were very proud of the successful completion of the long duration orbital flight.

DOCKING OF SOYUZ 10 WITH THE SALYUT ORBITAL LABORATORY

Space is ceasing to be an arena for occasional investigations. In space we now see systematic investigations of artificial Earth satellites, the accomplishment of more remote journeys by unmanned interplanetary spacecraft, and constant useful and necessary operations of meteorological and communication satellites. The scientific investigations and photography of geological and geographic items, continents and bodies of water in various regions of the Earth's surface make it possible to develop techniques for using the data obtained in industry. Study of the pictures of the Earth's surface and natural features in different spectral regions of the electromagnetic spectrum makes it possible to obtain valuable information as to the nature of the Earth's surface, geomorphological characteristics of an area, and information for industry and agriculture regarding soil, crop conditions, and much else. The prediction of ore deposits, which therefore facilitates search for useful raw materials, is also possible using unmanned spacecraft. Unmanned spacecraft can also give information about ocean currents, the level of water contamination, the level of ocean swell, accumulation of plankton, and the state of the ice in polar latitudes.

By means of a spacecraft, we can measure the Earth's self-radiation as a hot body, which results in information not only about the Earth's surface temperature, but also concerning the state of the atmosphere. Wide use is made of astronauts' observations of the state of the Earth's cloud cover; these help in detecting the point of origin of cyclones, typhoons, and

hurricanes, and in tracking their movement. For geologists, geographers, and geodesics and other specialists who study the planet Earth, the launching of manned and unmanned spacecraft into space has opened up areas previously inaccessible for survey. In general, the perception of the most important features of terrestrial relief, soils, and plant cover from space is improved, not only by the great distance from the planet, but as has been shown by the specialists, by the optical thickness of the atmosphere. This thickness perturbs astronomers who observe the planets and the stars, but in these global observations, it conceals the finer secondary details of the picture which passes in /180
front of the objective of a camera mounted on a spacecraft.

Pictures of a single locality taken with a small time interval between them make it possible to determine these stereoscopically, thus generating a three-dimensional picture of the surface geometry and cloud cover over large areas. Another point is interesting: in order to see better what is taking place deep in a planet, one needs to go higher in an Earth orbit. The useful minerals of different kinds are deposited in quite specific geological structures, which have typical special microrelief features and drive the basic rock up to the surface. This makes it possible to see regions where useful minerals are deposited.

The presence of a trained specialist onboard a spacecraft substantially increases and widens the circle of tasks which can be solved by cosmographic methods, in particular, reloading of equipment, changing of filters, choice of information for fast transmission to Earth, fine tuning and adjustment of instruments, and choice of the spectral range for photography (photography in the invisible infrared makes it possible to see things not visible to the eye).

All our knowledge of the planets, stars, galaxies, and the interplanetary and interstellar media have been obtained by astronomers from the emission of these objects. However, not all radiation passes through the Earth's atmospheric layer, but only rays of visible light, part of the thermal (infrared) radiation, and some of the radiation at radio wavelengths. The ultraviolet, x-ray, and gamma radiation of space objects is inaccessible to terrestrial observatories. In addition, the atmospheric air layers are always in motion, and their optical properties are always changing. Therefore, the launch of equipment into space is the beginning of a new phase in the development of astronomy. Now it has become possible, for example, to photograph the other side of the Moon and to measure the magnetic field of Venus. Our astronauts have repeatedly made meteorological observations, and these are being taken simultaneously by automatic systems. For example, Shatalov observed a strong cloud vortex associated with a strong cyclone, formed over the Atlantic. Volynov observed a storm over a large area of South America; over the Indian Ocean, he saw a tropical cyclone in the formative stage. Sevast'yanov observed a tropical storm on the daylight side of the Earth at a time when the waves of a coastal storm were also visible. These reports from our astronauts were used in making up forecasts in the weather service.

The second decade in the history of spaceflight opened with the launch of Soyuz 10 (Figure 72). The successful launch on April 19, 1971, of the orbital laboratory Salyut (Figure 73) was an important step in the conquest of space.

The building of the Salyut laboratory and its successful launch into Earth orbit shows that the Soviet Union is accomplishing its planned program of investigation and conquest of space, in particular in creating long term scientific laboratories in /181

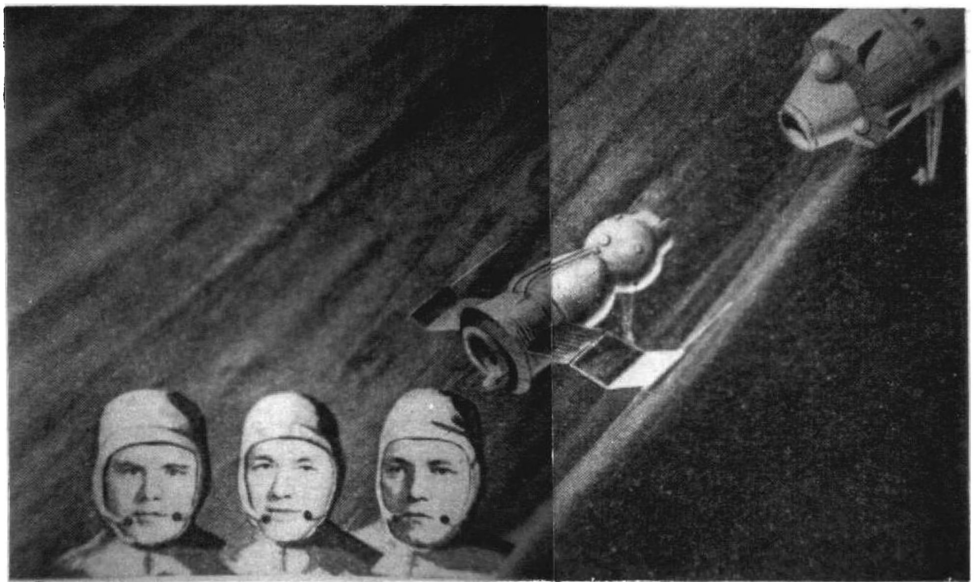


Figure 72. V. A. Shatalov, A. S. Eliseyev, and N. N. Rukavishnikov, who accomplished flights in the Soyuz 10 spacecraft (during the flights, Soyuz 10 rendezvoused with the Salyut orbiting laboratory and docked with it).

orbit, which will make it possible to make space investigations at higher levels, in conjunction with unmanned spacecraft, and to obtain continuous regular scientific information in the field of space physics, astronomy, and astrophysics, and the biological sciences, and thereby to assist with very complex scientific, engineering,medical and biological experiments.

When Salyut began its seventieth orbit around the Earth, a new launch was prepared at the Baikonur launch site. At dawn on April 23, 1971, the Soyuz 10 spacecraft, with a crew composed of the twice winner of the Hero of the Soviet Union award, V. Shatalov, A. Eliseyev, and the experimental engineer N. Rukavishnikov, was launched into orbit. Shatalov, Eliseyev, and Rukavishnikov carried out a set of scientific and engineering investigations in the joint flight with the first scientific Salyut laboratory and made a comprehensive check of the sophisticated onboard systems of the spacecraft, continued the development of manual and automatic control systems for control, attitude control, and

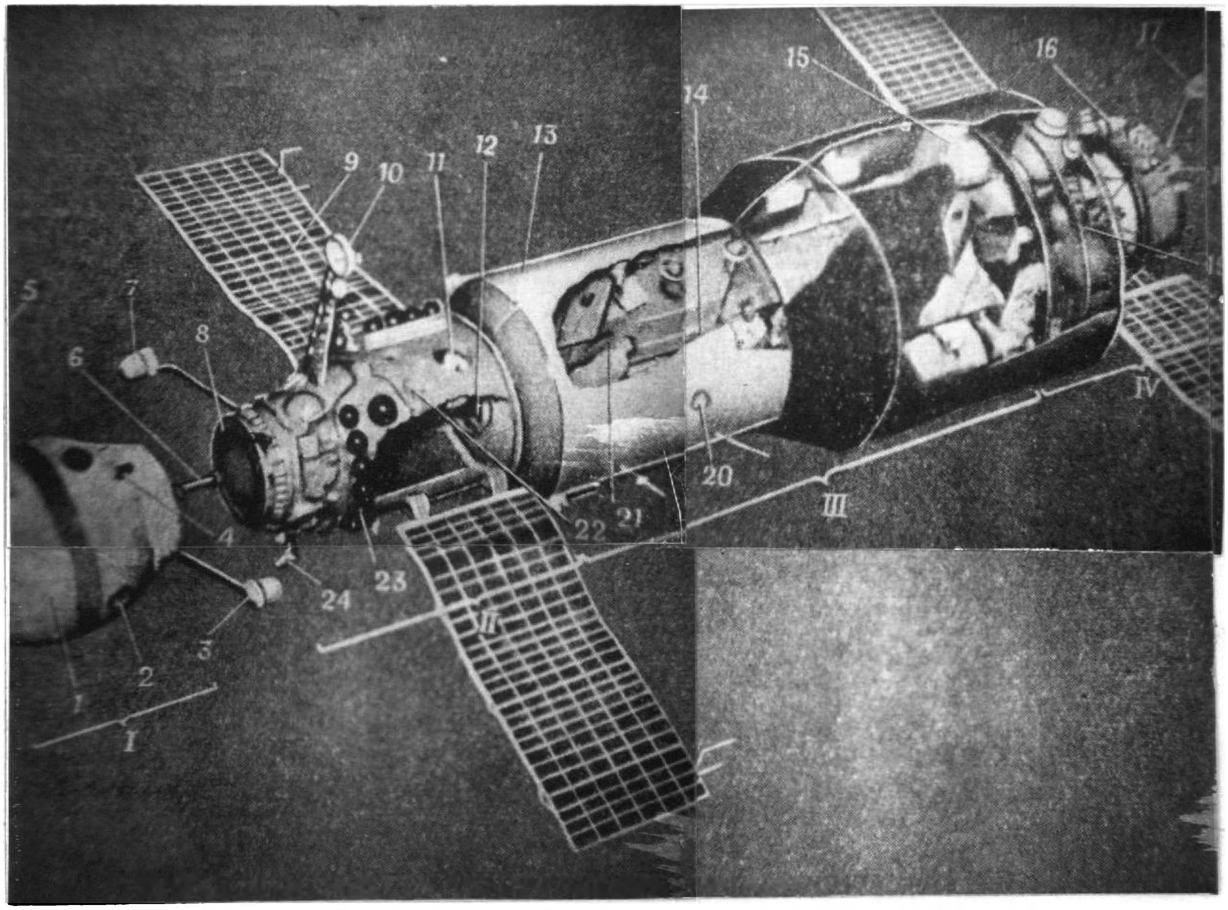


Figure 73. The Salyut orbiting laboratory.

I- the Soyuz spacecraft; II- transfer module; III- working module; IV- equipment module; 1- orbital compartment of the Soyuz spacecraft; 2- windows of the orbital compartment; 3- survey antenna of the search (spacecraft) system; 4- television camera; 5- antenna of the guidance (spacecraft) system; 6- stub of the spacecraft docking mechanism; 7- survey antenna of the search (laboratory) system; 8- conical receptacle for the docking device (laboratory); 9- solar cell panels; 10- antenna of the homing (laboratory) system; 11- the Orion astronomical system; 12- the hatch between the transfer and working compartments; 13- light meters; 14- scientific equipment of the working compartment; 15- equipment for everyday use; 16- micromotors of the attitude control system; 17- survey antenna of the search (laboratory) system; 18- external survey television camera; 19- systems of the equipment compartment; 20- windows of the working compartment; 21- astronaut's couch and laboratory control panel; 22- windows of the transfer compartment; 23- spherical tank with compressed gas supply; 24- attitude control sensor.

stabilization in different flight regimes, and also conducted medical and biological investigations of the effect of spaceflight factors on the human organism.

On April 24, the flight control center reported that, during the joint flight of Soyuz 10 and the Salyut laboratory, scientific and engineering experiments were conducted, together with docking and undocking exercises of the manned spacecraft and the Salyut laboratory. During these exercises, the astronauts verified the principles of approach and mooring of the spacecraft and the laboratory, and operated new docking devices, as well as a complex of radio engineering equipment.

A very complex technical experiment for search, approach, and docking of the manned spacecraft and the unmanned laboratory began in the early orbits of the flight. The astronauts carefully prepared Soyuz 10 for its meeting with Salyut. Aided by the ground stations, they carried out an approach to a distance of 180 m from the laboratory. Further approach and mooring was done by the crew, and on April 24, Soyuz 10 was docked with the laboratory. For 330 minutes, the first flight of the very first space system consisting of a spacecraft plus laboratory in the docked condition continued. The crew of Soyuz 10 used the time to check the onboard apparatus and to evaluate the dynamic characteristics of the system. Following completion of the planned experiments, Soyuz 10 was undocked and separated from the laboratory. Having accomplished their program of experiments, the spacecraft crew began to prepare for the final stage of their space session. The spacecraft was placed in a new attitude appropriate for the departure from orbit mode and two hours into April 25, the retroengine was fired. Separation of the spacecraft modules took place, and the entry capsule entered the atmosphere at the design angle and performed a soft landing on target, several hundred kilometers from the Baikonur launch site.

In their two days, the crew accomplished a number of scientific and engineering experiments and investigations and conducted the first work with the orbiting scientific laboratory.

The people of the Soviet Union are pleased by and proud of their country's new scientific and engineering success, and proud of their scientists, spacecraft designers, technicians, and working groups and organizations who participated in preparing and carrying out the launch of the Salyut laboratory and Soyuz 10 spacecraft.

LONG TERM ORBITING LABORATORIES /184

On June 6, 1971, the Soyuz 11 spacecraft was placed in Earth orbit. A new stage of operations with the Salyut laboratory began. The crew of Soyuz 11, consisting of G. Dobrovol'skiy, Vo. Volkov, and V. Patsayev, carried out a smooth docking with the Salyut laboratory (Figure 74). After docking and performing the required system tests, the astronauts transferred from the spacecraft to the laboratory in order to begin a large program of scientific and engineering investigations and experiments.

The experience gained by astronautics indicates that the use of spacecraft in Earth orbit opens up very great possibilities for solution of urgent practical problems in diverse areas of man's activity on Earth.

Orbital laboratories hold out very great promise for the accomplishments of programs of this kind. Further space conquest is evidently impossible without constructing large orbiting complexes with spacecraft bases and transport ships for servicing systems consisting of several specialized space items, both manned and unmanned, to carry out a large volume of diverse investigations, including the study of:

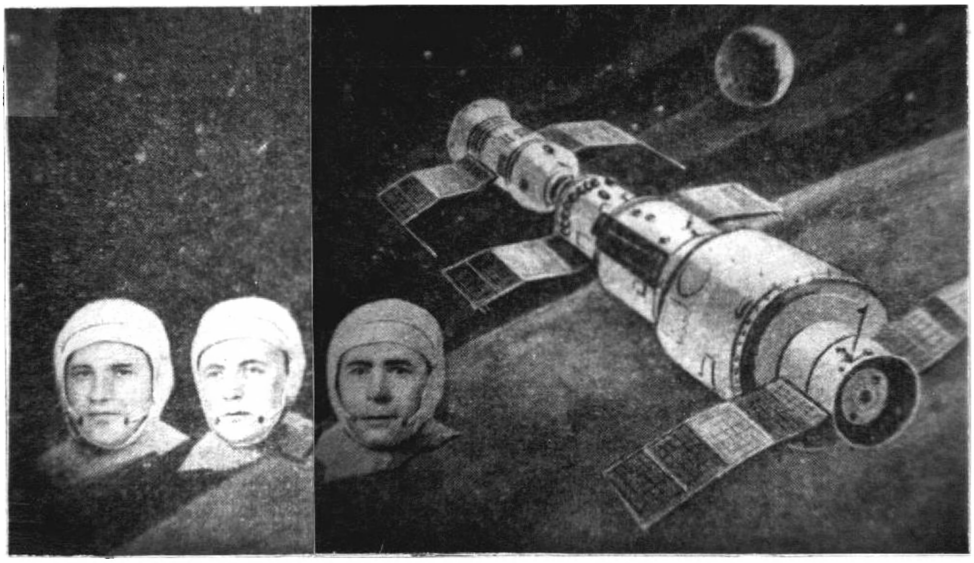


Figure 74. G. T. Dobrovol'skiy, V. N. Volkov, and V. I. Patsayev, who made a long duration flight in the Salyut laboratory and the Soyuz 11 spacecraft.

1) the Earth and the space around it, to obtain valuable data in the field of meteorology, oceanology, geophysics, which would be of great value to industry; in particular, at present, photographic data sent back from space are used to study the geological structure of the Earth and to make weather forecasts;

2) the effect of space conditions (this cannot be determined on the Earth), i.e., the effect of a combination of weightlessness and vacuum on the accomplishment of special physical and chemical experiments, and the provision of conditions unobtainable on Earth for special technical processes of great practical interest;

3) possibilities of future use of space laboratories as large energy systems capable of converting solar energy and transmitting it to Earth to supply energy to ground facilities.

Orbiting laboratories could be used in the future as a platform for assembly and launch of interplanetary spacecraft. They open up great possibilities in the field of astrophysics investigations which are perturbed by the Earth's atmosphere. It is a complex scientific and engineering problem to create orbital complexes.

First, it is necessary to decide how to ensure conditions for the subsistence and operation of a human in space. This means, initially, that one must provide the crew with food and water, and maintain the necessary atmospheric composition in the compartments. In the first stages, it will be possible to do this by providing supplies sent from Earth by means of transport ships. Later on, the problem of ensuring life support for humans will evidently be solved by creating a closed cycle for materials onboard the laboratory.

Initially, this job will be done by the so-called semi-closed ecological systems which provide complete regeneration of the atmosphere and water in the laboratory. Then closed systems will be built which provide a complete recycling of the materials. These will use low-order plants of the chlorella type and higher order plants, and also animals and birds to furnish fresh meat products onboard the laboratory. However, the problem of creating conditions for normal human life during a long duration space flight involves not only supply of air, food, and water, but also furnishing conditions in which the astronauts can withstand the weightless state over a long period. One cannot exclude the possibility that one will have to create an artificial gravity force on board the laboratory during long spaceflight of humans, since one cannot as yet answer the question, can human beings live and work in the weightless state during a flight duration of several months?

The mechanics for creating artificial gravity force are quite clear: the laboratory should rotate about its center of mass. Because of the centrifugal acceleration then arising, a force could be created equal to that of gravity acting on the human body on Earth, or even exceeding it. However, it is as yet not clear how the human being would respond to this. It is also not known whether or not the constant rotation of the laboratory would disturb the observations on the Earth and the heavenly bodies. /186

One cannot exclude the possibility of creating a special space laboratory, a self-contained unit of the orbital system, in which an artificial gravity force is created by continuous rotation, and which flies in close formation with the base laboratory. The crew could then live alternately in this laboratory, and in the main laboratory. In this case, one would require reliable equipment, systems, and assemblies with considerable operational spares. One should also provide for possible equipment repair in flight by the crew of the subsidiary laboratory.

The development of orbital space laboratories is closely connected with the task of creating multiple use transport systems. Here we are concerned with space systems in which a transport spacecraft would be capable of multiple flights from Earth to the space base and return.

Complex technical problems arise also in creating an onboard complex necessary for flight and attitude control of the orbiting base laboratory in space, and to create special equipment and instruments for conducting a program of scientific investigations and for resolving economic problems. In particular, astrophysical investigations require large space telescopes with a specific resolving power. Special optical systems and radio electronic

equipment, and special onboard cryogenic systems are also required for orbiting laboratories which are to operate for a long period in space.

In the Soviet Union, these tasks are performed in stages. The result of the flights of the Cosmos series and the Soyuz series was the development of a set of systems for rendezvous and docking in space. This has made it possible to construct the first experimental laboratory in Earth orbit and to obtain experience in a lengthy (18-day) spaceflight. The world press has given wide publicity to this new achievement of our country.

The world press has noted that the creation, by Soviet scientists and technicians, of orbital stations and constantly operating orbital laboratories, observatories, and workshops with specialist crews heralds the onset of the era of actual systematic use of deep space for the needs of science and industry.

The press, radio, and television media in the Socialist countries have given very high praise to the boldness of the project, and the accuracy in the use of the Soviet Union's new space experiment.

The creation and successful operation of this station has /187
been appraised as an outstanding feat of Soviet astronautics, and as evidence of the great success in the development of Soviet science and technology.

The first manned Salyut orbiting laboratory was made up of the following basic modules: the transfer, working, and equipment modules.

The transfer module is one of the inhabited compartments of the station, designed for the conduct of scientific observations and experiments. It consists of the docking unit for docking the station with a spacecraft in artificial Earth satellite orbit and for transfer of the astronauts from the spacecraft to the station and back again. The body of the module is sealed. The module contains elements of the thermal control, life support system, scientific equipment, control panels, and, on the outside, it has the homing antennas, light gauges, ion sensors, the external survey television camera, solar cells, units of the thermal control system, bottles of air, and the Orion telescope. The inside surface of the compartment is covered with a decorative finishing coat. The transfer module communicates with the working module by means of a lock, equipped with both automatic and manual actuators.

The working section of the station is sealed. It serves as working and living quarters. It contains the basic instruments and systems. During the periods of station control, conduct of scientific investigation and observations, the members of the crew occupy this compartment. Here they take their food and rest. The body of the working module consists of two cylindrical shells: an upper shell (adjacent to the transfer module) of 2.9 m diameter, and a lower shell of a little over 4 m diameter, joined to the upper shell by means of a conical adapter. The module contains supplies of water, food, equipment of the life support system (regeneration units, carbon dioxide absorbers, and units for making and heating food), the radio engineering and television apparatus, the apparatus for control of the onboard units, electrical supplies, attitude and motion control, telemetry, panels, and working places for the crew members, the interior structural members, equipment to assist the crew in moving around and in staying in a fixed location,

and scientific and experimental apparatus. The interior surface of the compartment is covered with a decorative finish. On the outside of the working module, there are radiator panels of the thermal control system, communication antennas, radio telemetry antennas, solar sensors, and viewing devices.

The plant compartment houses the engine module, the correction motor, the module of attitude control motors, protective screens for the engine unit and the auxiliary equipment systems. The support skirt for this compartment has mounted on it antennas of the radio communication system, antennas of the orbit radio control system, an illuminating lamp, units of the thermal control system, and the ion sensors. The motor module brings together into one entity the units of the auxiliary equipment system and the correction motor. It also contains tanks with propellant for the auxiliary actuator systems, the /188 external survey telecamera, spherical supercharge tanks, service panels for the system of auxiliary actuators and the solar cell panels.

The attitude control motor module contains the main and the duplicate micromotors for control of pitch, roll, and yaw. The outer surface of the module carries the survey antennas of the search system. There are about twenty illuminating lamps, both demountable and fixed, for operation with attitude control and navigation instruments, and photography and visual observations in the transfer and working modules of the laboratory.

The mass of the Salyut system is: Soyuz 25 tons, length 2.14 m, with a maximum diameter of module of 4.15 m, and the free volume of the sealed compartment of 100 m^3. The mass of the laboratory without the spacecraft is 18,900 kg. The solar panel area is 42 m^2.

The astronauts Georgiy Dobrovol'skiy, Vladislav Volkov, and Viktor Patsayev spent 23 days onboard the scientific laboratory. The unparalleled space watch of the very first manned scientific station in Earth orbit was the focus of widespread interest. Mankind followed the Salyut flight with great interest. By brilliantly accomplishing their assigned program, the crew of the Salyut-Soyuz space system wrote a new page in the history of the conquest of space; this was a logical step in peaceful progress, an important stage towards uncovering nature's secrets in order to use her riches in the interests of man.

The flight performed by this crew showed that Soviet science and technology had successfully solved the basic technical problem of creating space stations in Earth orbit. By creating the first manned orbital laboratory, the Soviet Union was able to make a deep and comprehensive study of questions of first rank importance for science and industry.

The creation of the Salyut-Soyuz space system stands alongside similar outstanding accomplishments, like the launch of the first artificial Earth satellite, the flight of Yuriy Gagarin, the spacewalk of Aleksey Leonov, the return of the lunar soil sample, and the operation of the lunar rover (Lunokhod). It would be difficult to exaggerate the contribution which astronauts Dobrovol'skiy, Volkov, and Patsayev have made to the development of manned scientific laboratories, and to the conquest of space. Their names will be written in gold letters on the pages of history. They died at their post in the interests of science and a bright future for mankind. Their loss will not be forgotten. The death of the first three to open up space does not stop people's desire for knowledge. To think otherwise would be to offend against their memory, to give up against the forces of nature, and to arrest the course of history.

"Astronautics is a young science, and when one sails in the stellar ocean, one cannot yet be fully protected from unforeseen events." These words of Yurii Gagarin were carried by many foreign newspapers on the day when the whole world was shocked by the death of our space heroes. Ahead of us, there are new flights, new achievements, and the creation of new manned laboratories. /189 They will be larger and more complex, multi-purpose and specialized, but the feats of the Soviet astronauts will never be erased from man's memory.

The crew of Salyut took a spectral picture of the Earth's surface (in the Caspian region), photographed cloud formations above Povolzh'ya; carried out medical and biological investigations, tests of new instruments for attitude control using the Sun and the planets, and also space optics instruments; carried out experiments using the gamma telescope, the onboard astrophysical Orion observatory and the multi-purpose Era apparatus; observed the development of various plants in the weightless state ("the space kitchen garden"); took navigation measurements using the onboard computer; measured the level and the tissue dosage of radiation, which is of great practical importance for increasing the efficiency of the dosimetric control system; and constantly observed the formation of typhoons and cyclones, and reported them to the ground service.

The flight of Salyut, which lasted for roughly six months, is an important new step on the road to create long term orbital laboratories in Earth orbit.

It consisted of several stages. The first stage was the simultaneous flight of the laboratory along with the Soyuz 10 spacecraft. At this time, a check was made of the function of the sophisticated systems for search, docking, and undocking of

the spacecraft and the laboratory. The next work in the operation of the orbital laboratory was performed in the automatic mode. The second stage began on June 6, 1971, by the launch into orbit of the Soyuz 11 spacecraft. Following successful docking with the Soyuz 11 transport craft, the laboratory became the first manned orbital scientific laboratory.

After the completion of the manned flight, the laboratory again continued in the automatic mode. During the subsequent flight of the Salyut laboratory in the automatic mode, constant scientific and engineering investigations were carried on and control was maintained over the operation of systems, units, and sceintific apparatus of the laboratory under conditions of long term residence in space.

During the whole flight, the onboard equipment of the laboratory operated normally. The temperature and pressure in the sealed compartments was maintained within the assigned limits.

On October 11, 1971, the retroengine was switched on, and the laboratory transferred to a descent trajectory, entered the atmosphere above an assigned region of the equatorial Pacific Ocean, and its lifetime came to an end. The valuable experimental data obtained by Salyut will be used in the design of new space systems, with which it will be possible to observe from space
the state of the atmosphere, clouds, and snow cover, to obtain /190
constant information regarding the boundaries of the ice cover in the oceans and the seas, concerning the thawing of icebergs, concerning floods in rivers and other natural changes, to observe the places of origin of cyclones, hurricanes, and typhoons, and to observe their development and migration, and to make a substantial improvement in weather prediction.

Space laboratories can study questions relating to Earth resources in a new way. They can give information about the nature of the Earth surface and about rocks lying near the surface from spectral studies of sections of the surface in the ultraviolet, visible, infrared, and microwave regions of the electromagnetic spectrum. This information is very valuable, both in a scientific and a practical sense. The thermal characteristics of the Earth's surface, ocean currents, places of accumulation of plankton and fish - all of this can be studied and has already been studied in part from space. As has been mentioned, the Earth's atmosphere absorbs shortwave electromagnetic radiation, in particular spectral sections, and the Earth's ionosphere reflects radiowaves coming from space and from the Sun over a wide range of frequencies. In addition, atmospheric fluctuations create a great deal of noise in the study of many at astronomical features, but this noise does not affect the accuracy of apparatus mounted in orbital laboratories. The study of the x-ray and ultraviolet solar radiation and of the solar corona will allow a thorough evaluation of the effect of solar activity on processes occurring on Earth and in the Earth's atmosphere, and will also facilitate the prediction of solar flares, an important matter for improving the safety of remote manned spaceflight and of long term residence of astronauts aboard orbiting laboratories.

Medical and biological experiments have been performed, starting with the first manned flight in space, and will continue steadily, since only an experimental check of the effect of various flight conditions on the human body will allow proper planning of an expedition for distant and long duration space flight. This relates to the capability of the human, the development of life support systems, the rationale of food, flight comfort, and the optimum combination of human activity

and automatic devices in flight control. It will clearly be necessary also to check the activity of humans in emergency situations to avoid trouble and when spacecraft systems have to be repaired. Scientific and technical experiments under vacuum conditions have already been conducted in part by our astronauts as regards welding of metals, but it is quite clear that deep vacuum and the weightlessness effect opens up very great promise for the development and accomplishment of many technical processes onboard orbital stations for microelectronics, electron beam technology, for matching of crystals, the obtaining of super-pure materials and for much else which is impossible or very difficult to accomplish in terrestrial conditions. Orbital laboratories will clearly be used also as orbiting launch sites for interplanetary spacecraft, as space bases for assembly of /191
spacecraft, and repair and testing of all their systems before undertaking long duration flight.

The conquest of space is a task requiring very great skill, energy, courage, and will power. It is clear that the Seventies will be the years for development and wide application of long-term manned orbital laboratories with exchange of crews. This will allow scientists and engineers to go from occasional experiments in space (flights of the Vostok, Voskhod, and Soyuz type, as well as the flights of the American spacecraft Mercury, Gemini, and Apollo) to a regular space watch. The Soviet Salyut laboratory is the first such station.

With the object of further study of space, the Soyuz 12 spacecraft was launched on September 27, 1973, in the Soviet Union, manned by a crew consisting of the commander, V. G. Lazarev, and the engineer, O. G. Makarov. Their orbital flight program, designed for a two-day period, included:

— a set of checks and tests of sophisticated onboard systems;

— further development of manual and automatic control processes in different various flight regimes;

— conduct of spectrographic studies of individual sections of the Earth's surface, to obtain data relevant to agricultural problems.

The flight program was accomplished, and the crew landed successfully in the assigned area.

The Soyuz 13 spacecraft was launched on December 18, 1973, with a crew of P. Klimuk and V. Lebedev. After accomplishing their mission, the crew made a safe landing on Earth.

SECOND GENERATION UNMANNED SPACECRAFT

The successful development of space science and technology allows tasks of increasing complexity to be solved in the study of space near the Earth and near the Moon and in studies of planets of the Solar System. In 1966, the first spacecraft made a soft landing on the lunar surface. This was the unmanned Soviet spacecraft Luna 9. Its investigations were important for understanding the processes occurring on the Moon. The first direct determination of the physical and mechanical properties of the Moon's surface layer was made by the unmanned spacecraft Luna 13. New data concerning the Moon's surface layer were obtained by means of artificial satellites.

In principle, a new stage in the study of the Moon was opened up by the launch and successful accomplishment of a complex flight program by the Soviet spacecraft Luna 16. For the first time in the history of astronautics, an unmanned spacecraft returned to Earth samples of lunar soil, after accomplishing the Earth-Moon-Earth journey. This flight clearly demonstrated the possibilities which unmanned spacecraft and control systems were opening up for the study of the Universe. The construction of the automatic Lunokhod 1 and the investigations for the first time in history by unmanned devices opened up the possibility of performing scientific experiments not only at the landing site, but also at locations at some distance from it. Self-propelled lunar vehicles (Figure 75) can perform chemical and mineralogical analyses of lunar soil along the itinerary of the vehicle, determine the hardness, and transmit the information gathered to Earth. Using lunar vehicles, one can investigate a wide expanse of territory, not only studying the physical and chemical composition of the mantle, its mechanical and other properties, but also obtaining a large volume of information concerning the structure of the lunar surface, and can transmit to Earth pictures of the surrounding landscape. Lunokhod 1, which landed on the surface of the Earth's natural satellite on November 17, 1970, from the unmanned spacecraft Luna 17, was the first automatic laboratory designed for a sophisticated study of the structural features of the lunar surface, the medium surrounding the Moon, and remote space objects. As a result of the information obtained, topographical pictures along the Lunokhod's itinerary were constructed, local maps of sections of greatest interest in the topographical and geological sense were drawn, and a topographical description of the test sections was made. The topographic study of the locality was made by taking photographs of the lunar landscape, including both panoramic and fixed photographs, and also telemetry data along the path traversed, measurements of the heading, roll and pitch attitude

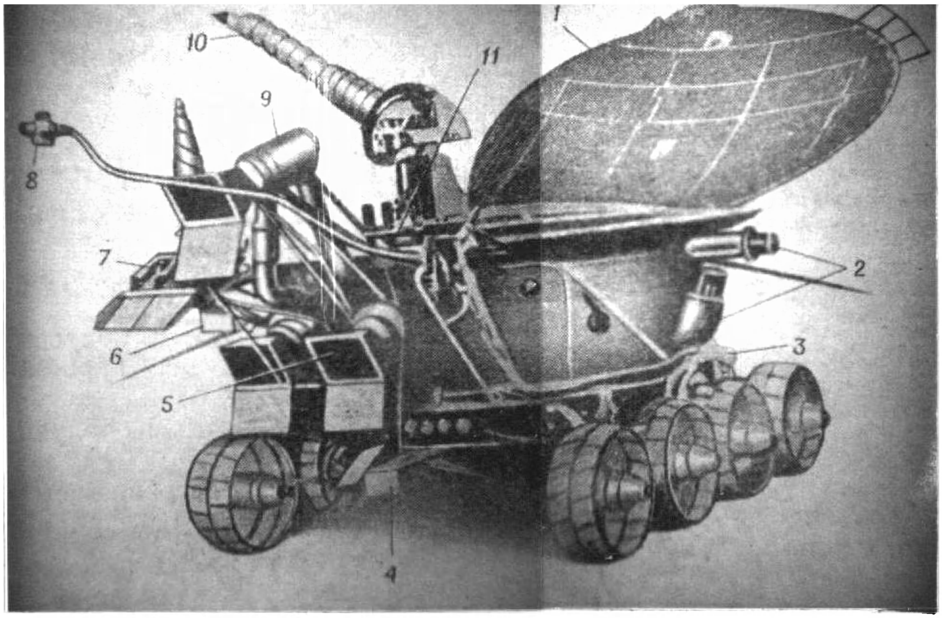


Figure 75. The Lunokhod 2.

1- solar cell; 2- telephotometers; 3- instrument for conductivity determination; 4- extension unit of the "Rifma" apparatus; 5- television cameras; 6- astrophotometer; 7- corner reflector; 8- magnetometer; 9- extension television camera; 10- narrow directional antenna; 11- optical detector.

of the Lunokhod during the motion. During the photography, spectroscopic panoramic views were obtained of several craters, allowing their structure to be studied. The physical and mechanical properties of the soil were studied also from pictures of the tracks of the Lunokhod wheels on the television views. From the depths of the wheel marks and the nature of the deformation of the soil under the wheels of the Lunokhod while in straight-line motion and while turning, estimates were made of the strength properties of the soil along the entire path traversed by the vehicle.

The scientific program of Lunokhod 1 also included the first test experience of using the Moon for study of individual regions of the Universe by means of an x-ray telescope. X-ray radiation includes radiation from individual sources and a diffuse

background from space, which can be very useful for astrophysical observation of the Universe. Thus, the setting up of an x-ray telescope on the Moon marks a new stage in the development of extra-atmopsheric astronomy. The Lunokhod also carried radiometric equipment to measure the flux of solar and galactic cosmic rays and to monitor the radiation environment during the flight of Luna 17 to the Moon and during operation of the Lunokhod vehicle. An important scientific experiment in the work program of Lunokhod 1 was the conduct of a laser-location exercise by Soviet and French scientists, using a special reflector mounted on the Lunokhod and designed for accurate measurement of the distance bewtween the Earth and the Moon.

The Luna 17 spacecraft consisted of a unified landing stage and an automatic traveling laboratory, the Lunokhod. The landing stage is a self-contained rocket unit, whose basic tasks are: to carry out trajectory correction on the Earth-Moon leg; to inject the spacecraft into orbit around the Moon; to generate a pre-landing lunar orbit, and to land on the lunar surface. The landing stage of the spacecraft includes the Lunokhod and the folding ramp for it to reach the surface. The automatic self-propelled Lunokhod 1 has a mass of 756 kg, and consists of two basic parts: the instrument section and the wheeled chassis. The instrument section is sealed, and made of magnesium alloy. The upper part of the body is used as a radiator-cooler in the Lunokhod thermal control system and is covered by a special lid which performs two functions: during the first lunar night, the radiator closes down and prevents radiative loss of heat from the module; during the lunar day, it opens, and the solar cells, mounted on its outer surface, charge the batteries which supply the onboard electrical equipment. The Lunokhod electrical supply /194 system consists of a solar and chemical battery. The main source of electrical energy is the solar cell. Its actuator is controlled from the Earth.

The forward part of the instrument section contains lamps for the television cameras and an electrical drive for the steerable narrow directional antenna, which is used to transmit television pictures of the lunar surface to Earth. The television systems of the Lunokhod are designed to:

a) transmit television pictures of the locality to Earth, these being necessary to control the Lunokhod motion from Earth;

b) obtain a panoramic picture of the surrounding locality and photograph sections of the sky, the Sun, and the Earth, in order to obtain an astronomical fix for the Lunokhod.

On the left and on the right, the Lunokhod carries a telephoto camera and antennas for receiving radio commands from the Earth. An isotropic thermal energy source is used to heat the gas which circulates inside the equipment. Alongside it, there is an instrument to determine the physical and mechanical properties of the lunar soil. The instrument compartment also has instruments of the Lunokhod electronic control system and the onboard radio system, which receives radio commands from the Lunokhod control center and transmits information from it to Earth.

To maintain the required thermal conditions on the Lunokhod, a special system was developed and installed which must react to the sharp changes in temperature when day and night interchange on the Moon's surface and when there is a large difference between temperatures of parts of the equipment located in sunlight and in shadow.

The instrument compartment is mounted on an eight-wheeled chassis, which provides a high degree of mobility over the lunar surface and reliable operation of the Lunokhod over a long period of time. The self-propelled chassis provides motion with two

forward and reverse speeds, and rotation in place and while moving. All control of the motion is accomplished using five command motions and the "stop" command. The system for safe motion provides automatic stop of the Lunokhod for limiting angles of roll and pitch.

Control of the Lunokhod is maintained by a special crew from the remote space communication center. The crew consists of a commander, a driver, a navigator, and an engineer.

Choice of movement is made on the basis of evaluating television information and the operational incoming telemetry data about the roll and pitch, concerning the path traversed, the state of the wheel drive and their operating conditions. With a distance of about 400 thousand kilometers between the /195 control center and the controlled object, the transmission time for a radio signal from the Earth to the Moon and back is about 2.6 seconds. Therefore, if one thinks of the time required to evaluate the information obtained and to conceive solutions regarding further actions, the time interval between events requiring a change of activity and the actual change is about four to six seconds, which naturally complicates the Lunokhod control.

On October 4, 1971, a program of scientific and engineering investigations was accomplished, as developed for the very first lunar unmanned self-propelled Lunokhod 1 equipment. The successful operation of this laboratory continued for ten and a half months. During this time on the lunar surface, in conditions of space vacuum, radiation, and considerable variations in temperature, and a complex surface topography along the path of motion, all the systems and the scientific instruments of the self-propelled vehicle functioned normally, thus allowing

accomplishment of both the main and supplementary programs of scientific investigations of the Moon and of the space environment, as well as programs of engineering and structural tests.

While conducting these investigations and tests, the Lunokhod traveled a distance of 10,540 m, which allowed a detailed mapping of the lunar surface over an area of 80,000 m^2. The television systems of the Lunokhod obtained more than 200 panoramic and more than 20,000 still photographs of the lunar surface. At more than 500 points along the Lunokhod path, studies were made of physical and chemical properties, and at 25 points, an analysis of the chemical composition of the surface soil was made. The simultaneous development of results of television photography of the surface, telemetered information, and measurements of the physical and mechanical properties of the soil and its chemical composition have allowed a quantitative evaluation to be made of the topographic and morphological features of the lunar surface in the regions of the Lunokhod operation. The multi-month operational time of the Lunokhod has permitted long duration and planned measurements to be made of cosmic x-ray radiation and a study of the radiation environment on the Moon to be made. The active operation of the Lunokhod ended because of decay of the isotopic heat source, which led to a fall of temperature within the equipment during the eleventh lunar night (from September 15 to 30, 1971). Using the French corner laser reflector mounted on the Lunokhod and directed towards Earth, a determination of location from the Earth will be made for several years. The widespread information obtained using the Soviet Lunokhod 1 vehicle will serve to increase our knowledge of the Moon, the Sun, and of cosmic space.

Figure 76. The Venera 7 interplanetary spacecraft (drawing).

On the eve of 1971, when Lunokhod 1 was transmitting its unique information regarding the lunar surface, the Soviet spacecraft Venera 7 (Figure 76) accomplished a landing on the surface of the planet Venus. For the first time in the history of Earth /196 astronautics, Earth received simultaneous information from the surface of two different heavenly bodies.

Venus has attracted the attention of scientists for several centuries. However, traditional investigations of it by means of terrestrial telescopes, as has been pointed out, meets a natural barrier: Venus is constantly hidden by a dense layer of cloud. Comparatively recently, new information on the planet has been obtained from dioastronomical and radar investigations. But such basic characteristics of the planet as its chemical composition and atmospheric pressure had not been determined at all.

The unmanned spacecraft was launched to Venus from an intermediate Earth orbit on August 17, 1970.

The basic objective of the launch of Venera 7 was to make a landing on the planet, to study the atmosphere during the descent right to the "bottom," and to make measurements directly on the surface. In addition, during the interplanetary part of the flight, a study of the intensity of cosmic rays was planned.

During the flight, two trajectory corrections were made, to ensure arrival of the spacecraft at the planet at a time of radio visibility from Earth measuring points. After a flight of 120 days, during descent into the planetary atmosphere, the /197 descent module separated from the orbital module. The entry aerodynamic forces reduced the speed of the descent capsule relative to the planet from 11.5 km/sec to 200 m/sec. During this period, the loading reached 150 g, and the temperature of the shock layer between the shock wave and the spacecraft body reached 11,000° C. On the day of the landing, the distance between Earth and Venus was 69.6 millions of kilometers. The radio signals transmitted by the spacecraft reached Earth after 3 minutes 22 seconds.

The Venera 7 spacecraft consisted of a descent module and an orbital module; the total mass was 1180 kg. The descent capsule was designed for an outside pressure of up to 180 atm and a temperature up to 530° C.

To measure the pressure and temperature of the atmosphere of Venus, the Venera 7 descent module carried a special apparatus to measure temperature in the range from 25 to 540° C, and pressure from 0.5 to 150 atm. The atmospheric parameters of the planet surface at the landing point were: temperature 475 ± 20° C, pressure 90 ± 15 atm. Thus, the flight of Venera 7 gave direct

measurements of the atmospheric parameters right down to the surface. It was finally established that Venus has an exceedingly strongly heated atmosphere, whose surface density is greater than that of the Earth by a factor of about 60.

Inside the sealed instrument compartment of the entry capsule, there was radio engineering, telemetry, and measuring apparatus, automatic and power supply units, and also a fan for the thermal control system. The top part of the entry capsule, above the instrument section, contained the parachute system. The parachute cupola was designed to operate at temperatures up to 530° C.

The orbital module was designed to carry the entry capsule to Venus right down to atmospheric entry. The entry capsule was attached to the upper part of the orbital section, and its lower surface carried the correction motor, while on its side surfaces there were solar cell panels, the radio command antenna, optical instruments, and auxiliary devices of the star attitude control system.

The electrical supplies for the spacecraft were derived from chemical current sources, and the electrical energy stored in accumulators was supplemented by the solar cell current. An attitude control system was carried, as well as a programmed time device and other units.

In the design of the entry capsule, provision was made to protect it from the very high temperatures and pressures occurring during entry into the atmosphere of the planet, during parachute descent, and after landing on the surface.

The scientific results transmitted by Venera 7 have considerably widened our knowledge of this planet.

On February 14, 1972, Luna 20 was launched. After 105 hours /198 of spaceflight, the spacecraft was placed in orbit around the Moon. On February 19, a correction was made to achieve landing at the designed point on the Moon, and on February 21, the spacecraft made a soft landing in a mountainous area.

After landing and checking the state of the onboard systems, the phototelemetric equipment was switched on. The pictures of the lunar surface obtained with this equipment allowed a choice of location for taking soil samples. The soil sampling mechanism carried out borings and took in samples of lunar soil, which were then transferred to a container in the Earth return section and sealed. On February 23, the Luna 20 space rocket with the return capsule, using the landing stage as a platform, was launched from the Moon and, on February 25, it approached the Earth with the second cosmic speed. The re-entry capsule separated from the rocket. Thereafter, the re-entry capsule was tracked by ground-based radar stations during its flight until the landing on the Earth. When the re-entry capsule entered the atmosphere, it was decelerated aerodynamically, and, in the final section, the parachute system was deployed, and the re-entry capsule completed its landing in the designated area with great accuracy. With the flight around the Moon of Luna 9, which accomplished a soft landing on the lunar surface, the hardness characteristics of the lunar soil became known for the first time. The first information on the physical and mechanical properties of the lunar soil was obtained by the unmanned spacecraft Luna 13, which carried instruments to determine the hardness and density of the soil. Lunokhod 1 opened up wide opportunities for choosing and investigated the most interesting local areas of the lunar surface. Luna 16 returned

friable soil samples from the maria regions of the lunar surface to the Earth for laboratory investigation. Similar soil layers were returned to Earth also by the Apollo spacecraft. However, almost all the information on lunar soil obtained by spacecraft refer to the maria regions which constitute a small fraction of the lunar surface. A considerably greater part of the Moon - its back side, the polar regions, and the continental areas of the visible surface - remain little known.

We know least about the lunar rock making up the continents. Investigation of the material of these areas would make it possible to answer a great many questions connected with the history of the formation of lunar soil. The flight of the Luna 20 spacecraft, which returned soil from the continental areas to Earth, was a very important step in the study of our nearest space neighbor.

On March 27, 1972, the Venera 8 spacecraft was launched. The main task for this new space experiment is to continue the investigations of the planet Venus accomplished by spacecraft. In July, 1972, after flying a trajectory of 312 million kilo- /199
meters, Venera 8 reached the vicinity of the planet. The program called for separation of the descent capsule, which accomplished a smooth descent into the atmosphere of Venus and carried out scientific measurements.

During its flight on the Earth-Venus leg, the spacecraft carried out investigations of the physical characteristics of interplanetary space, and, in particular, measured concentrations of neutral hydrogen and the solar plasma flux.

CHAPTER 7

HISTORY OF ASTRONAUTICS ABROAD /200

FLIGHTS OF THE MERCURY AND GEMINI SPACECRAFT

In the USA, manned spacecraft were developed first in the Mercury program. The result was construction of a one-man spacecraft designed to study the possibility of human flight in space and the part of the human in control of a spacecraft, as well as to investigate the effect of space conditions on a man. Four Mercury spacecraft carrying astronauts were placed in geocentric orbit. Later, in the Gemini program, two-man spacecraft were built, whose function was to study the possibility of long term flight (up to 14 days) of humans in space, and the possibility of encounter in orbit of manned and unmanned spacecraft and of a space excursion by astronauts. Twelve spacecraft of Gemini type were placed in Earth orbit in the period 1964 — 1966, using the Titan 2 launch vehicle. The Gemini spacecraft had a control system for its translational motion. The spacecraft provided emergency ejection of the astronaut's couch and a hatch. In contrast with the first American spacecraft, Gemini had a control system for rendezvous in orbit and docking with the Agena launch vehicle. Figure 77 shows the spacecraft layout in the form in which it re-entered the atmosphere. Each of the emergency hatches had a window to give vision at encounter and docking in orbit, observation of the horizon when the retromotor was fired, and of the Earth's surface at landing. The spacecraft commander had manual control of the attitude control system with his right hand, and the second astronaut (if need be) — with his left hand. The second astronaut in this spacecraft controlled the

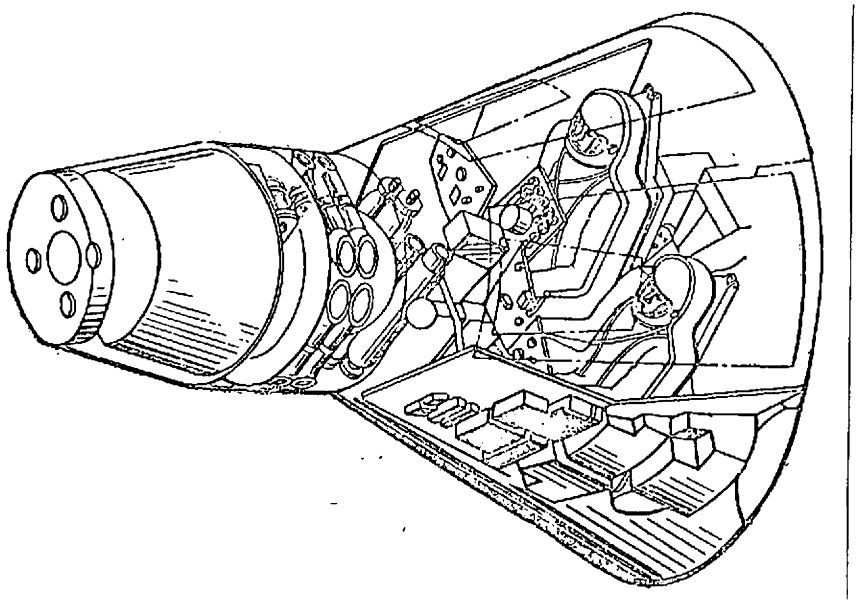


Figure 77. Layout of the Gemini spacecraft.

operation of the computer, the propellant elements, and the passive spacecraft (the Agena rocket module). Either one of the astronauts could give a command to operate the ejection couch during an emergency landing. During landing of Gemini under normal conditions, a parachute system would operate, as was true also for the Mercury spacecraft. In the active part of the flight, the spacecraft commander would initiate the emergency ejection, since he had control of the launch vehicle. Regardless of who gave the command, both emergency hatches would be jettisoned simultaneously and, thereafter, both couches would eject at once. The spacecraft had an emergency rescue control with which the four retromotors would be simultaneously ignited. They would achieve complete separation of the spacecraft from the launch vehicle at great height where the aerodynamic drag was small. The Gemini spacecraft, in contrast with its Mercury predecessor, was equipped for conducting experiments in orbit. Gemini used a modular construction for its systems and all the elements of each system were located in a compact group. This

allowed any system to be dismantled without the other systems being disturbed. To make best use of the cabin volume, the pilots sat side by side, but each couch was turned by twelve degrees with respect to the other. The cabin had only indicator meters, controls, and the elements of the life support system.

The electric energy source on the spacecraft for flight of more than two days duration was hydrogen-oxygen fuel cells (they are lighter than silver-zinc batteries). Storage of the gaseous oxygen and hydrogen for use in the fuel cells, and gaseous oxygen required for breathing would entail facilities with unreasonably high mass. Therefore, a decision was made to use a cryogenic method of storage of oxygen and hydrogen.

For maximum operating reliability of the spacecraft, there was complete duplication of the jet control system. This is used only for attitude control during operation of the retromotor and for stabilization and control during atmospheric re-entry.

The major part of the flight in Gemini, as a rule, was conducted without communication with Earth. But the ground facilities were arranged so that communication was possible with at least one facility once per revolution of the spacecraft around the Earth.

For rendezvous in orbit and docking with another spacecraft, the vehicle was equipped with radar facilities, onboard computers, and motors capable of small accurate maneuvers. Two-component self-igniting propellants were chosen for the control motors to provide rendezvous in orbit and atmopsheric re-entry.

The conical part of the spacecraft carried a special radar unit. This provided information on the headings, range, and closing speed of the spacecraft and the Agena rocket, and also gave dial readings of the measured values of range and closing speed.

The spacecraft carried an inertial navigation system, consisting of a computer and a stabilized platform. Using a special unit, the astronaut could feed data to the system from the computer.

Special devices were provided for docking of Gemini and Agena. Through the flat window in the conical part of the re-entry capsule, the astronauts could observe the whole docking process without distortion. During docking, the first contact of the spacecraft was made with a floating cone attached to the forward part of the Agena rocket by means of a shock absorption suspension. This cone absorbed the docking impact energy in any possible direction, eliminated a bounce effect and simultaneously steered the spacecraft towards the spring-loaded locks. After docking, the mechanism rigidly clamped the cone to the Agena rocket, combining the spacecraft and rocket into a single structure and making it possible to maneuver the two together in space.

Before undertaking a maneuver, the astronauts had to check the operation of the main rocket systems. Therefore, the Agena rocket systems contained several parameters which were measured and read to indicate the state of the rocket systems. The readings were displayed on an instrument panel located outside the rocket in such a way that either of the astronauts could see them, both before and after docking.

To carry out a brief experimental excursion of an astronaut into space, the entrance hatch was not bolted down, but was held by mechanical drawbars (used when an astronaut made an excursion into space in his special suit). /203

The problem of landing the spacecraft entailed control of flight during atmospheric entry and during landing. Control was achieved using a lift force, controlled by displacing the spacecraft center of gravity, and varying the roll. The onboard inertial navigation system and the computer generated the necessary control signals. Twenty astronauts took part in flights in the Mercury and Gemini program.

THE LUNAR SPACECRAFT

The Apollo spacecraft was used by the USA for flights to the Moon, landing on the Moon, and return to Earth. The crew consisted of three astronauts. The launch vehicle used was the Saturn 5, which injected a spacecraft mass of more than 40 tons into a lunar trajectory.

The launch vehicle placed in a flight trajectory to the Moon the spacecraft consisting of three modules: the crew module, the equipment module, and the lunar module. During the flight of the spacecraft to the Moon, the equipment module motor provided the necessary braking to inject the spacecraft into selenocentric orbit. Two astronauts transferred from the crew module to the lunar module, which then separated. Its motor povided the retardation required to transfer the spacecraft into a lunar approach and landing orbit. When the astronauts were ready for the return trip from the Moon, the lunar excursion module motor provided the launch from the Moon, injected the spacecraft into orbit around the Moon, and provided the rendezvous with the

crew and equipment modules. The astronauts transferred from the lunar module to the crew module, and the lunar module separated and remained in selenocentric orbit. The crew module motor injected the crew module into the Earth trajectory. The equipment module remained attached to the crew module until the beginning of Earth re-entry. Here the equipment module ensured atmospheric entry of the crew module at the appropriate angle and then separated.

In choosing the spacecraft form for rendezvous in lunar orbit, the USA adopted the principle of a spacecraft made up of individual modules, so that the principle of "efficient weight" could be used by separating spent elements. The arrangement of the spacecraft modules is shown in Figure 78.

The crew module is the only module which returns to Earth, i.e., performs atmopsheric entry and landing. Its form is decided by the fact that it is the main center of the spacecraft. All operations are accomplished by the astronauts within this module, and it, therefore, contains the communication, navigation, control, computing, and indicating equipment. The forward part /204 of the module carries the Earth landing system, the jet control motor system, the emergency system, and the forward hatch. The rear part carries motors and the propellant tanks of the jet control system.

The crew module has five windows. Three of these provide a view of the forward hemisphere, necessary for the general observation and for docking in orbit with the lunar excursion module; two (side) windows provide a view during launch and when the horizon meters are used.

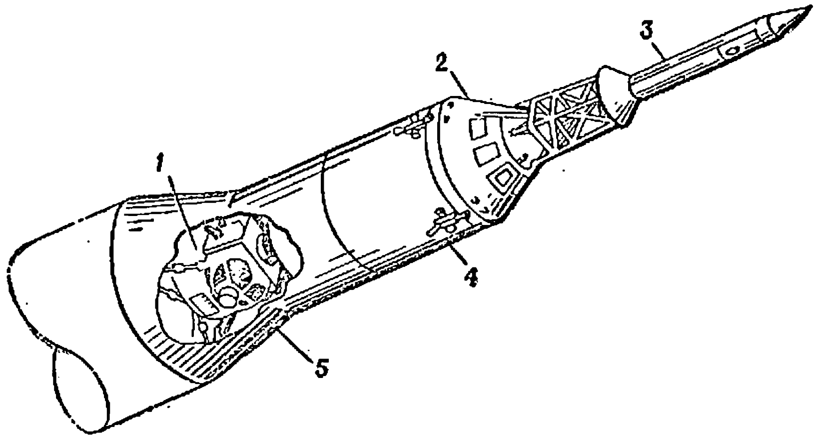


Figure 78. The Apollo spacecraft.
1- lunar excursion module; 2- crew module; 3- emergency rescue system for the spacecraft in the launch phase; 4- the equipment module; 5- the transfer module.

The crew module has a heat shield for atmospheric re-entry protection, and also protection during injection into orbit is provided by a heat shield covering of ablation material. During the atmospheric re-entry, the surface temperature reaches 2760° C, as reported in the foreign literature. However, because of the efficiency of the ablative material, the temperature at the rear of the heat shield and in the stainless steel internal structure is quite low.

To ensure the most favorable conditions during atmospheric re-entry, the Apollo crew module has a lift to drag ratio of about 0.5, and a cone angle of 33 degrees.

Control of range during atmospheric re-entry is accomplished by varying the roll attitude.

It should be mentioned that, during flight, the crew has no communication with the equipment module and, therefore, this module contains equipment and systems which do not require servicing and direct control and which are not required following separation of the crew module from the equipment module. The /205
equipment module, as has been mentioned, is jettisoned before atmospheric re-entry.

The emergency rescue system for the Apollo spacecraft was developed to a considerable extent on the previous spacecraft. It consists of a tower structure with a motor to separate the crew module and a motor for separating the tower. The system also includes an auxiliary motor to control the pitch angle. In case of necessity, it can pull the spacecraft away to one side of the launch vehicle flight trajectory. All the motors are solid fuel. The tower is a welded structure of titanium tubing and is mounted on four supports attached to the crew module by four explosive bolts.

In the case of a launch vehicle explosion on the launch pad, the emergency rescue system lifts the crew module to a height of about 1200 m, at which point, the landing system comes into operation.

The spacecraft navigation and control system is self-contained and consists of a computer, a three-axis stabilization platform, and an optical system to correct the inertial system during flight. Using the optical apparatus, the astronauts can measure the angle between directions to a star and to the Earth with great accuracy, or to the Moon, to determine the spacecraft location. In the case of malfunction of the navigation and control system, corrections can be made using data from Earth trajectory measurement and computers, which can be transmitted along with the appropriate commands to the spacecraft over the two-way communication line.

Attitude control and stabilization of the spacecraft was accomplished using a group of sensors which develop signals proportional to the angular speeds of the spacecraft, a set of accelerometers which sense the acceleration along the spacecraft longitudinal axis, and a set of gyroscopes which determine the spacecraft angles relative to the three axes. The system must be capable of manual control relative to the three axes to generate small changes of velocity during encounter and docking in orbit. In the middle section of the trajectory, control of the thrust vector is required. During atmospheric re-entry, control is accomplished by roll orientation of the spacecraft when at a large angle of attack. The jet control system consists of a large number of small liquid fuel rocket motors. On commands from the attitude and stabilization control system, these generate impulses to stabilize a given spacecraft attitude, change its position, and damp oscillations which can arise when aerodynamic forces act during atmospheric re-entry.

The jet control system for the crew module is used only /206
after separating it from the equipment module prior to atmospheric re-entry. It provides three-axis control in the period before large aerodynamic moments arise, control in roll, and damping of angular velocity of yaw during atmospheric re-entry.

The jet control system for the equipment module generates all the impulses required for attitude and control and stabilization of the spacecraft in all sections of the flight, apart from the time when the main equipment module motor is operating. The system also generates impulses for flight trajectory correction in the middle section and an emergency retro-impulse during orbital flight. It consists of four independent identical systems, each of which contains four control motors.

The air conditioning system is located in the crew module and provides cabin pressure stabilization and the required gas composition, circulates air through the cabin and the spacesuits, provides thermal control of the crew module systems, and injects and circulates air into the entry capsule after it enters the atmosphere.

The communication system consists of a two-way radio [illegible] telephone, an onboard system for outer trajectory measurement, an onboard system of telemetry measurement, and television equipment. The outer trajectory measurement system includes a space survey apparatus and a radar responder, used to measure range during trajectory measurements for orbit rendezvous. The onboard telemetry measurement system has to transmit, from the spacecraft to ground stations, parameters of the spacecraft system, parameters of the crew life support system, and the results of scientific experiments. The television system can be used for wideband television transmission, remote reception, control and observation of hazardous regions of the flight, at docking, and also to obtain documentary records of a flight process.

The motor unit of the equipment module is located in that module and provides a velocity correction impulse and an emergency impulse following separation of the rescue system. It provides flight trajectory correction in the middle section, reduces the speed at transfer to a selenocentric orbit, transfers the spacecraft from a selenocentric orbit to an Earth trajectory, provides the retro impulse for injection into orbit around Earth, for maneuvers at orbital rendezvous, and during emergency rescue, before or after injection of the spacecraft into the trajectory to the Moon.

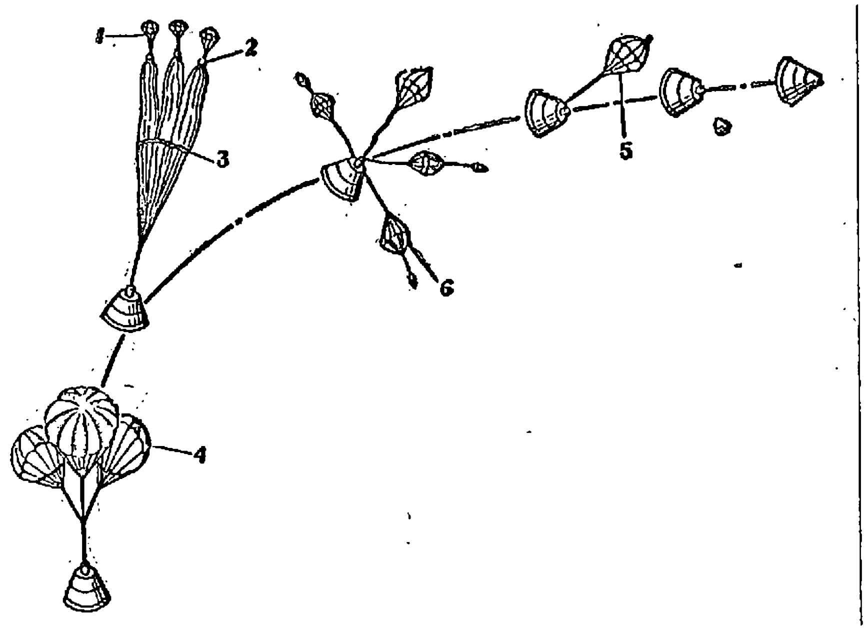


Figure 79. The spacecraft landing system.

1- draw parachute; 2- attachment point of draw parachute to main parachute; 3- main parachutes (drawn out); 4- main parachutes (fully opened); 5- release of braking parachute; 6- release of draw parachutes.

The Earth landing system provides a safe landing of the crew and the crew module over water or land. Figure 79 shows the sequence of operations carried out by means of the landing system. The crew module parachute system consists of a braking /207 prachute, three annular draw parachutes, and three main parachutes. The rate of descent with three parachutes is 8 m/sec, and with two, it is 10 m/sec.

The landing system begins to operate at a height of 8300 m, triggered by a barometric sensor, but the astronauts can give manual commands and observe the operation of the system through the window.

The Apollo spacecraft is designed for flight with maximum use of the capabilities of the crew. The crew consists of a pilot, a navigator, and an engineer.

The pilot occupies the control post on the left couch, in front of the main instrument desk, and controls maneuvers necessary to move the lunar excursion module, for orbital rendezvous, and Earth return. The navigator occupies the middle position. He is responsible for the operation of the navigation and control systems, and assists the pilot with control in critical stages of the flight. The engineer occupies the right hand couch. He assists the navigator and ensures operation of the equipment, whose readings he monitors from the indications on the instrument panel. When the pilot and the navigator are in flight in the lunar excursion module, the engineer operates the crew section module on his own.

The spacecraft can carry out the mission without receiving information and commands from the ground facilities. But this does not eliminate the possibility of using information received /208
from Earth, to increase the reliability and accuracy of all the observed parameters.

Launch of the first three models of the Apollo main unit (without crew) was performed in 1966 during flight tests of the Saturn IB launch vehicle. It was planned to use this launch vehicle in February, 1967, to launch the main Apollo spacecraft with three astronauts. However, on January 27, 1967, the spacecraft and three astronauts were burned during a fire arising at the time of ground tests.

Thereafter, a series of changes was made in the crew cabin, including a new exit hatch, which could open in 2 — 3 seconds (the hatch in the previous structure opened in 90 seconds); teflon insulation in the cabins was replaced by metal foil; the aluminum tubing was replaced by stainless steel tubing. These changes increased the spacecraft mass and increased the parachute

system. By reducing the spacecraft lift, the major part of the problem associated with the mass change was solved. Thus, a change in the structure of the entrance hatch without changing the aerodynamic lift required ballast of 136 — 180 kg to be carried. The solution adopted was to reduce the spacecraft lift to drag ratio from 0.35 to 0.28, which allowed the ballast mass to be reduced to 13.6 kg. This increase in the Apollo spacecraft mass also required a change in the structure of the parachute system, the cupola area of the braking parachute was increased, and two stages of opening were provided for the three main parachutes. Before the launch of the first Apollo 11 spacecraft, which landed with the astronauts on the Moon, several launches were carred out to test the lunar, command, and auxiliary (service) modules. Then manned flights were conducted in Earth and lunar orbits.

THE MOON LANDING

The flight of Apollo 11 with a Moon landing was accomplished from June 16 to 24, 1969.

The spacecraft mass was 43,860 kg, and the total payload mass of the Saturn 5 rocket was 49 — 71.5 tons, this comprising, in addition to the spacecraft, the emergency rescue system and the transition stage between the launch vehicle and the spacecraft. The spacecraft consisted of the main unit and a lunar cabin, containing the landing and flight stages.

For the launch, the Saturn 5 vehicle had a first stage with thurst of approximately 3470 tons, a second stage with a thrust of approximately 520 tons, and a third stage with a thrust of approximately 90 tons. The third stage and the spacecraft were injected into an initial, nearly circular, geocentric orbit. In

this orbit, the third stage and the spacecraft were oriented parallel to the horizon during auxiliary motors. Then, after a two-hour check of onboard systems, the second ignition was made to transfer from geocentric orbit into the flight trajectory to the Moon. Thereafter, the module configuration was changed. The main unit was separated from the transfer stage, joining it to the last rocket stage, moved ahead several meters, and rotated through 180 degrees. /209

After separation of the main unit, the transfer stage was jettisoned, and the main unit, now rotated, was docked with the lunar cabin, using the internal adapter. Then a check was made that the docking latches were fastened, and then the joints of the main unit and lunar cabin were brought together and the transfer tunnel and the cabin of the lunar flight stage were filled with oxygen (from the main unit supply).

After separation of the last stage, the auxiliary spacecraft motor was fired which gave it a velocity increment of 6 m/sec and removed it to a safe distance from the last stage before the remainder of the propellant was vented through the motor. As a result of the venting of propellant, the stage received a velocity increment and went into a trajectory from which it later transferred to heliocentric orbit under the influence of lunar gravity. On the Earth-Moon flight, up to four corrections were provided for. In this section, the spacecraft rotated about its longitudinal axis, and was positioned perpendicular to the Sun's direction, to avoid solar heating of specific parts of the body.

The spacecraft flight in selenocentric orbit and the landing on the Moon proceeded as follows. The auxiliary motor was fired to slow the craft before transfer to the initial elliptic

selenocentric orbit at a distance of 148 km from the lunar surface. When the spacecraft went behind the Moon and there was no communication with it, the motor gave the spacecraft a velocity increment change of 891.2 m/sec and transferred it to a new orbit. After two revolutions in this orbit, the auxiliary motor was fired again, and gave the spacecraft a velocity change of 48.1 m/sec, transferring it to an orbit which was to degenerate to the design base circular orbit of height 111 km under the influence of perturbations due to anomalies of the lunar gravitational field, at the time of rendezvous with the flight stage after its launch from the Moon.

Then one of the astronauts passed through the inside tunnel from the crew module to the lunar cabin, where he checked the operation of onboard systems for several hours, and returned to the crew module. Later, he again returned to the lunar cabin and was joined by a second astronaut. The two checked the operation of the onboard systems and began to prepare the lunar cabin for a separate flight and landing on the Moon, and first put on their spacesuits. After preparing the lunar cabin, they undocked it from the main unit, in which one astronaut remained. For thirty minutes, the two spacecraft performed a formation flight at distances of several meters apart. During this time, the lunar cabin was checked visually from the main unit, and then the main unit was separated from the lunar cabin, using the auxiliary motor, to a distance of 3.3 km horizontally, and 1 km vertically. Then the landing stage motor was fired and transferred the lunar module to an elliptical selenocentric orbit with a perilune height of 15 km and an apolune height of 111 km. At perilune, the lunar cabin was decelerated by means of the landing stage motor. Here three control methods could be used: automatic, semi-automatic, and manual.

In the remote approach stage, the flight height was 2.3 km, the distance to the landing point was 8.1 km, the horizontal velocity was 152 m/sec, and the vertical velocity 45.7 m/sec. If the astronauts saw a crater or rock which could present a hazard for the landing, they could maneuver to choose another landing point.

The radar system controlling the landing communicated data on the flight speed and height. The stage of close approach began. The flight height above the lunar surface was then 158 meters, the distance to the landing point was 550 meters, and the horizontal velocity was 20.4 m/sec. At this point, the cabin passed the critical height below which it would be impossible to perform an emergency return since the cabin would fall to the Moon in the time required to separate the landing stage and fire the main flight stage motor. The vertical descent to a height of 45 m was made, at which point the horizontal velocity was completely cancelled. The vertical velocity was maintained automatically at 0.9 m/sec. A capability was provided to hover and to fly horizontally, but then it would be necessary to continuously reduce the engine thrust, since otherwise the cabin, whose mass continually decreased because of expenditure of propellant, would begin to rise.

Immediately after the landing, checks were made of the onboard systems, so that if an irregularity was detected which would threaten the astronauts' safety, a slow emergency launch from the Moon could be made. After rest and sleep, the two astronauts donned their backpack life support systems, their gloves, jackets, and boots, and descended in turn to the Moon, carried out their planned observations, and gathered rock samples. After finishing their mission, the astronauts filled the cabin with oxygen and prepared for launch from the Moon. The main flight stage engine,

which was to launch them from the Moon and to inject them into an initial selenocentric orbit, had to impart a velocity increment to the stage of 1845.5 m/sec, and would use 2263 kg of propellant. Having accomplished the launch from the Moon, the flight stage completed the vertical ascent in a period of ten seconds, which ended with a velocity of 15 m/sec at a height of 76 meters. When the flight stage was injected into selenocentric orbit and reached a perilune height of 16.6 km, the Apollo vehicle was in front of it at a distance of 470 km. All the subsequent maneuvers /211
to achieve rendezvous were performed using the flight stage attitude control motor. The flight stage was injected into a near circular orbit of height 83 km, and then, by several firings of the attitude control system motor, it transferred into the orbit of the Apollo spacecraft and performed maneuvers to rendezvous with it. Following a flight in formation with the Apollo stage, they docked and the astronauts transferred from the flight stage to the crew module. The flight stage was jettisoned, and the Apollo vehicle used its auxiliary motors to withdraw to a safe distance. Thereafter, the main flight stage motor was fired, used up its propellant, and transferred the stage into heliocentric orbit. Then, after 135 hours 24 minutes and 34 seconds, the astronauts fired the auxiliary motor to transfer from selenocentric orbit to an Earth trajectory.

On the Moon-Earth leg, the crew module was separated from the motor module, re-entered the atmosphere, and performed a controlled descent using lift. It is mentioned in the foreign press that the nominal flight range in the atmosphere (from the point of entry to the touchdown point) was 2380 km. By control of aerodynamic lift, this distance could be varied in the limits of 2220 — 4630 km.

The food supply in the Apollo crew module was calculated on the basis of a meal three times per day. Part of the food was in dehydrated form, and before using, it had to be mixed with water; part was in gelatinous form, so that it could be eaten with a spoon; and part was packaged in ready-to-use portions. For drinking and mixing food, the astronauts used water formed in the fuel cells as a reaction product of the hydrogen and oxygen. In the crew module, the astronauts could take water from three taps, and in the lunar module — from one tap. The taps had special filters to trap hydrogen bubbles in the drinking water. First aid kits were available in the spacecraft and in the lunar module.

The flight program was completed, but because of unsuitable meteorological conditions (a storm), the landing on the Earth was made at some distance from the planned point.

THE FLIGHT OF APOLLO 15

We shall describe one of the subsequent Apollo lunar flights. This flight, part of the American Apollo program, was made in the period July 26 to August 7, 1971. It should be noted that the Apollo 13 mission was unsuccessful because of an explosion of an oxygen tank in the motor module, and, therefore, an emergency return to Earth was made instead of a lunar landing.

Apollo 15 (Figure 80) made a lunar landing at the eastern edge of the Sea of Showers, in the region surrounded by the Apennine mountains. Compared with previous spacecraft, Apollo 15 had a modified crew module, designed for a 16-day flight (previous modules were designed for 10.7 days). The mass of the module was increased to 900 kg. The mass of the modified lunar module was increased to 1200 kg. The cabin was designed for a lunar stay up to three days, in contrast with previous cabins,

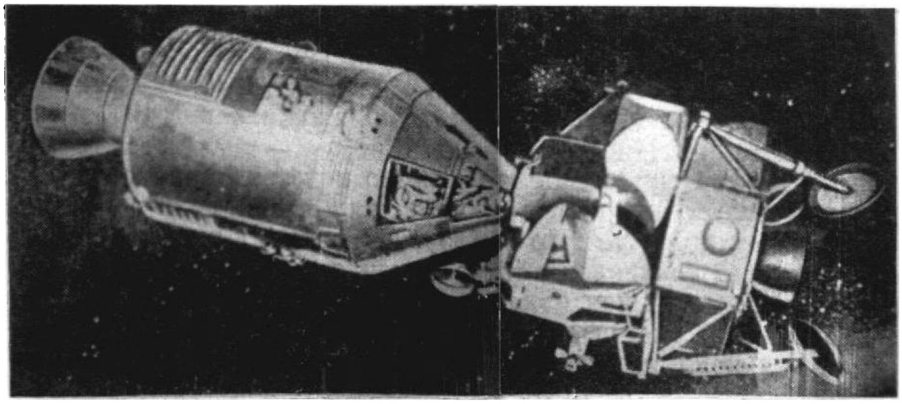


Figure 80. The Apollo 15 spacecraft.

designed for a lunar stay of 36 hours. The payload mass of the modified lunar cabin was increased from 270 to 450 kg, of which 200 kg was the lunar rover, which was not carried on previous missions. Thus, the mass of the modified Apollo stage was 30 tons, and of the lunar module — 16.3 tons.

The main module, as before, consisted of the crew module and the motor (auxliliary) module; the lunar module had the landing and flight stages.

The body of the crew module was of conical shape, made of layered panels (honeycombs and leaves of stainless steel), and had a heat shield made of honeycomb fiberglass and filled with ablative material (phenyl epoxy resin). The body of the motor compartment was a cylinder, also made up of layered panels, but differing from the crew module in that the honeycomb and leaves were made of aluminum alloy. To reduce the fire hazard in the crew cabin at launch, an oxygen-nitrogen atmosphere (60% oxygen) was provided. The astronauts used a self-contained life support system in their suits, and breathed pure oxygen. After injection into orbit, the oxygen-nitrogen mixture was expelled from the cabin and replaced by almost pure oxygen.

The astronauts' cabin had the guidance and navigation system. /213 The electrical supplies for the main module are hydrogen-oxygen fuel cells, located in the motor compartment. The electrical supply for the crew module, following separation of the motor module, is chemical batteries located in the crew module.

The main module has the propulsion motor, auxiliary motors, and the attitude control motors. The propulsion motor makes trajectory corrections in the Earth-Moon and Moon-Earth legs, makes the transfers to selenocentric orbit, corrects this orbit, makes a rendezvous with the flight stage (if need be), and transfers to the Earth return trajectory. The auxiliary motors, located inside the motor module, separate the spacecraft from the last launch vehicle stage, maintain three-axis spacecraft orientation, makes the trajectory correction on the Earth-Moon and Moon-Earth legs if a correction larger than the design value is needed (1.5 m/sec), and also separates the motor compartment from the crew compartment before atmospheric re-entry. The attitude control motors, located in the crew module, maintain the attitude of this module during atmospheric entry. This is required to give the spacecraft the ncessary roll angle, which allows it to control the lift during descent using the aerodynamic lift of the vehicle. The propellant supply system in all the motors is a forced one using helium.

The crew module is designed for controlled descent through the atmosphere using lift, with a terminal descent on parachutes. The parachute system consists of two braking parachutes, three draw parachutes, and three main parachutes. As is usual in such systems, the braking parachutes are deployed at a height of 7.7 km, reduce the speed of descent to 60 m/sec, and fix the module attitude for the draw parachutes to be opened at a height of 4.5 km, followed by the main parachutes, which give a module

splashdown speed of about 8 m/sec. The capsule attitude after splashdown in the upright position is maintained with the help of inflated floats (it has happened that the manned American spacecraft have turned upside down after splashdown).

The lunar module has two stages: the landing and the flight stages.

The liquid landing stage motor, with a variable thrust in the range 470 — 4470 kgf, performs the Moon landing and the main flight stage liquid motor performs the Moon launch. The flight stage has also 16 liquid attitude control motors with a thrust of 45 kgf each. These are used for attitude control, orbit correction, rendezvous, and mooring.

All of the motors of the main module and the lunar cabin operate with self-igniting propellant.

The lunar cabin has a guidance and navigation system. It includes a measuring unit, a computer, a television telescope, a radar for the lunar landing, and a radar for orbital rendezvous.

The energy supply in the lunar cabin is five silver zinc batteries, in the landing stage, and two batteries in the flight stage. /214

The skin of the lunar cabin consists of thin leaves of aluminum alloy. The astronauts' cabin on the flight stage is filled with pure oxygen, and has two hatches: an upper hatch communicating with the transfer tunnel, for transfer from the crew module, and a forward hatch for the lunar excursion.

The Apollo 15 spacecraft carried scientific instruments to investigate the lunar surface, and photograph and measure the Moon from the selenocentric orbit. The investigations on the lunar surface used seismometers, magnetometers, ion detectors, and instruments to determine the heat flux coming from the core of the Moon to its surface. In addition, samples of lunar soil were gathered, and laser corner reflectors were set up to allow experiments with observatories not equipped with high power telescopes. For investigations of the heat flux, probes were placed in deep holes (bored by the astronauts using electric drills). One of the sections of the motor module carried a group of equipment which included: a panoramic camera for photography from a height of 110 km, with resolution down to 3 m, a photogrametric camera, a spectrometer for determining chemical concentration of elements located on the lunar surface and below it, and other instruments.

The Apollo 15 astronauts traveled over the lunar surface using a rover with a total mass, including equipment and the astronauts, of 690 kg. The rover was a two-seat self-propelled vehicle (Figure 81), with which the astronauts completed three expeditions over the lunar surface, in a total time of six hours. During one expedition, there was malfunction of the front wheel steering, and the astronauts had to use the rear wheel steering system.

The equipment of the Apollo 15 astronauts was well thought out. The astronauts used suits of two types. The pilot of the main module had a suit with an inner lining made of knitted cotton with a quick release clasp from his belt to his neck, and attachments for fastening biosensors. The inner lining of the spacesuit had a layer of special "Nomex" material, a ventilated layer of nylon with a special covering, a sealed layer of nylon,

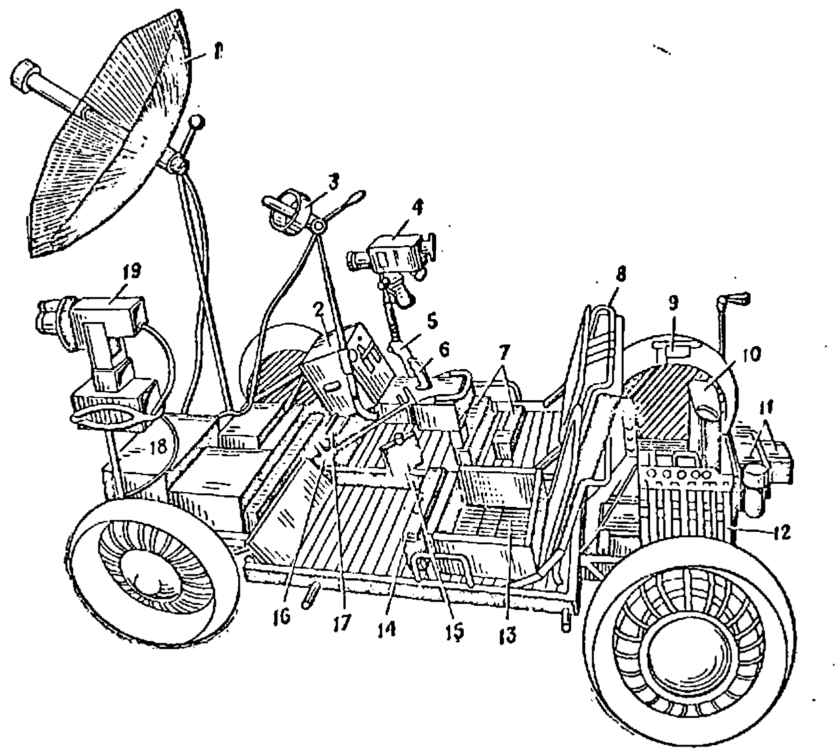


Figure 81. The lunar rover vehicle.

1- narrow directional antenna; 2- control panel; 3- low gain antenna; 4- movie camera; 5- handrail; 6- control stick; 7- cassettes and film; 8- hose; 9- electric drill; 10- stereoscopic camera; 11- magnetometer; 12- clamp for geological instruments; 13- containers for specimens (under seat); 14- container with brushes; 15- container with bags; 16- tongs; 17- place for camera; 18- receiver for direct Earth communication; 19- television camera.

and, finally, an outer covering composed of a layer of "Nomex," and two layers of glass cloth with a teflon coating. The spacesuit for the main module pilot was not designed for lunar excursion.

The lunar excursion spacesuits, besides the coatings common to the first type of suit, have additional heat protection and layers for meteoritic particle protection, consisting of two layers of nylon with special coating and several layers of glass cloth with a kapron coating. During excursion on the lunar

surface, the spacesuit lining next to the skin was replaced by "underwear" with water cooling. This "underwear" is made of nylon fabric and "Spandex" material, which is lined with plastic tubes for the water circulation.

In comparison with the suits for earlier flight to the Moon, the suits of the latter type were somewhat modified. They provided greater freedom of motion for the neck, shoulders, hands, waist, and legs; the mass of the Apollo capsule suit was 16 kg. The mass of the suit for lunar excursion, together with the self-contained backpack life support system, emergency oxygen supply, and frame with protective skin, was 93 kg.

The backpack system used on the Apollo spacecraft of later type had a mass of 64 kg and was designed for a period on the lunar surface of more than seven hours. It had a supply of breathing oxygen (oxygen was supplied under pressure), a supply of water for cooling the "underwear," a filter with lithium hydroxide to remove carbon dioxide in the oxygen circulation system, communication and telemetry equipment, meters, regulators, and a supply battery. The backpack has a heat insulating skin. As was the case for Apollo 14, the emergency oxygen supply, designed for 90 minutes, was located in two tanks on the backpack. The astronauts were equipped with a device which would allow the spacesuit to be switched from a malfunctioning life support system to the function system of the other astronaut, and thus to provide cooling of the malfunctioning suit, without using oxygen from the emergency supply for cooling. This special unit used a hose through which water from the functioning system went to the suit. A hose of 2.3 m length was carried on the rover. Two visors on the spacesuit helmet afforded protection from heat and light rays, and also from meteorite particles. Before going out onto the lunar surface, the astronauts donned

special gloves with heat insulating lining to protect their hands when operating with very hot or cold objects. The ends of the glove fingers were made of silicone rubber to increase the sensitivity. Under their helmets, the astronauts put on head phones with two sets of receivers and microphones.

Water was provided during the astronauts' period on the lunar surface. A small tank of water (225 g) was mounted on the inside of the helmet, and, to drink, the astronaut could turn his head and take water from a mouthpiece. The helmet also had a food capsule and a feeding tube which the astronaut could suck during the excursion.

In flight, the astronauts took off their spacesuits for long periods and wore "underwear," light teflon suits, consisting of shorts and shirts. The suits gave good heat retention and had pockets in which necessary items could be kept.

The daily ration of food for one person during the Apollo 15 flights was 2500 kcal. Part of the food, as in other flights, was dehydrated, and had to be mixed with water before use; part was gelatinous, and part was packaged in ready-to-use portions.

The Apollo 15 crew performed the following tasks: they returned to Earth 77.5 kg of lunar rock specimens; they set up passive seismometers on the lunar surface; they observed and photographed a number of Moon phenomena; they photographed 12% and measured 20% of the lunar surface from selenocentric orbit; they tested the Moon rover, the improved spacesuits, and the backpack life support system. However, during the flight, they observed many shortcomings and technical difficulties. In particular, one of the three main parachutes malfunctioned during the descent. This malfunction was very dangerous, since

it threatened the astronauts' life, because the uninflated parachute might have become entangled in the shrouds of the other two parachutes and prevented them from operating. The most probable cause of this malfunction was the fact that the nylon parachute shrouds were damaged when monomethyl hydrogen was released, which is capable of attacking nylon. This liquid was used on Apollo 15 as a fuel for the attitude control and stabilization system motor. Before the command module was realeased, its residue — 2.3 kg — was allowed to escape. During the landing, the astronauts reported that they observed the uninflated parachute through a cloud of monomethyl hydrazine vapor. Therefore, the decision was taken in subsequent Apollo flights not to jettison the residue of this fuel. Another cause for parachute failure could have been disintegration of the ring to which it was fastened. The faulty parachute was not fished out of the water, but one of the parachutes which functioned had a defective ring. Therefore, it was decided, in the future, to make the ring out of nickel-steel alloy, instead of steel.

The landing was made with two parachutes and, therefore, the vertical speed reached 9.7 m/sec instead of 8.5 m/sec, as with three parachutes, and the loading on impact was 16 g. The astronauts expected a difficult and hard landing, but all ended happily. The command module with the astronauts entered the atmosphere at a height of 120 km, with a speed of 10,989 m/sec.

The foreign press reported an interesting detail. Analysis of the rock returned to Earth by Apollo 15 shows that the age of one of the specimens is 4.15 billion years ± 200 million years. Scientists estimate the age of the Solar System to be 4.6 billion years.

It is known that the Apollo program ended with the flight of Apollo 17, from December 7 to 19, 1972. This spacecraft was planned to be the seventh and last American mission to the Moon. During this mission, the astronauts landed on the Moon and made three excursions on its surface to set up instruments and gather specimens of lunar rock. The journeys on the rover were longer than for the previous missions on the Moon and consistuted a total of 35.7 km. The total duration of the three excursions was 22 hours 5 minutes 6 seconds, compared with 21 hours in the previous flight. During this flight, 113 kg of specimens of lunar rock were gathered and returned to Earth and more than 2000 photographs were taken.

As a result of the lunar missions in the Apollo program, almost 400 kg of lunar rock specimens were returned to Earth and almost 15 km of photographic film were used in the panoramic and topographic cameras, as well as numerous photographic plates and movie film. Five groups of scientific instruments were set /218
up on the Moon. The total duration of human residence on the Moon in the Apollo program was 299 hours and 34 minutes and the duration of excursions was 80 hours and 14 minutes. All the launches from the Earth's surface, beginning with Apollo 8, were carried out with the Saturn 5 rocket. However, the foreign investigators view the results of the Apollo science program with restraint. In particular, they mentioned that the previous flights brought to light more new puzzles than they solved.

The lunar excursions have not made a qualitative jump in our fundamental understanding of the origin of the Moon and the Solar System, as they anticipated. The majority of specialists and reviewers consider that the USA will not send new missions to the Moon before the end of this century. One reason for this, which would make it difficult to restart such programs, is the disbanding of groups of experimental and highly qualified scientists and engineers.

According to accounts published abroad, there has recently been an investigation of the process of exciting electrical discharges in storms by passing a rocket through a cloud. The occasion for the investigations was the launch of Apollo 12, which took place in heavy rain, and the launch vehicle experienced a lighting shock. On the launch day, the following facts were recorded. At a time 36.5 seconds after the launch, lightning struck the launch vehicle while it was at a height of 2 km. At the same time, the cameras recorded a lighting discharge to Earth near the launch platform. The astronauts on the spacecraft reported that they saw a bright lighting flash, and one of them later reported that he felt a bump. In the command module, numerous emergency signal instruments showed emergency conditions, and the fuel elements were cut off from the main electrical supply rail. Then the air conditioning system did not operate for 60 seconds, and nine temperature and pressure sensors went out of action.

After 52 seconds from the end of the launch, when the rocket had reached a height of 4.2 km and was 650 m above the zero isotherm level, there was a second breakdown in the equipment, at which time the inertial unit was switched off. At that moment, no lightning was noticed, but the Earth-based radio receiving equipment for atmospheric discharge direction finding was saturated.

Subsequent analysis of the telemetry signals showed that electrical disturbances of smaller amplitude were seen, both before the two main breaks in the operation of the spacecraft equipment, and after them. However, for six hours before the launch and roughly the same time after it, no storm discharges

were recorded in the region of the launch site. Measuring instruments located in this area recorded only a somewhat increased electrical field intensity, which is typical of normal unstable weather. A counter located at the nearest air base recorded only a single storm discharge, the one which occurred 36.5 seconds after launch. All this indicates that the flight of the rocket through the electrified clouds created an electric discharge which would not have occurred under normal conditions. The cause of the discharge may be the fact that the metal body of the rocket, of about 100 m in length, and the jet of discharged ionized gas, of more than 200 m in length, promoted an increase in the electric field intensity along the rocket flight path by roughly a factor of 950.

The record of the electrical storm discharge on the camera film and appropriate calculations indicate that a charge of about 20 Kl passed through the rocket case in about 0.12 seconds. Thus, the assumption is that the observed lightning belonged to the long-duration lightning type. The investigation reports that the electrically conducting channel along which the lightning propagates can reach a temperature of 30,000° K, that there is almost complete ionization of the air, and that the density of free electrons available for electrical conduction is about $10^8 m^{-3}$ 50 ms after the start of the discharge. Such a long duration retention of conductivity is due to the slow recombination of electrons during cooling of the channel to about 3000° K. The temperature of the gases discharging from the nozzle of the rocket engine is less by a factor of 5 or 6 than in the channel and, therefore, the density of free electrons in the jet, as well as their lifetime, are considerably less than in the lightning propagation channel. One can assume that, as early as some tens of milliseconds after combustion of the propellant, the density of free electrons in the jet is practically zero, and the cessation of luminosity of the jet is

a sign that there are no free electrons in it. However, the rocket leaves behind it a long trail of gases with high concentration of large and small ions, the total charge of which is evidently non-zero. The presence of a large number of ionized particles in the jet of discharged gases, although it does not provide such a conducting channel for the lightning as do free electrons (since the mobility of ions is more than an order of magnitude less than for free electrons), nevertheless, affects the path of the electrical discharge, acting in the direction of propagation of the lightning leader (the channel for propagation of lightning leading to the Earth is called the lightning leader). If the mechanism for propagation of the lightning leader is photoelectric ionization of particles ahead of it, then the ionized wake of gases can serve as a path along which the lightning leader can reach the Earth. The rocket body and the trail of ionized gases, of total length as great at 600 m, is in essence a long conductor extending into the electrical field of the clouds. Such a conductor causes a concentration of electrical lines of force, which result in the breakdown voltage being exceeded and a corona discharge occurring. If the electric field is strong enough, the corona discharge converts into lighting.

The role of the rocket in forming lightning depends also on the electrical charge accumulated in the cloud. In order to maintain an electrical discharge and form lightning, the amount of electricity in the cloud must be large enough, as otherwise the discharge will extinguish. The energy of natural lightning ranges from 10^8 to 10^{10} joules. An important factor may also be accumulation of charge on the body of the rocket when there is no external electrical field, which can arise in various ways. The amount of the electrostatic energy can be calculated from knowledge of the electrical capacity of the rocket body. However, it should be noted that the maximum charge that can

accumulate on the rocket body is negligibly small compared with that carried by lightning. At the same time, the electrical charge of the rocket body can promote initiation of lightning during the flight of the rocket through an electrified cloud. The authors of these investigations conclude that the lightning stroke observed during the launch of Apollo 12 confirms the need to protect electronic equipment, particularly the computers of the rocket guidance system, from voltages associated with lightning.

To ensure safety during launch of a rocket through clouds, it is necessary:

1) to take into account that the launch is less hazardous if the cloud thickness does not exceed 1700 — 2500 m;

2) to carry out a radar scan to evaluate the increase of reflections, which is a measure of the convection and electrical activity in clouds;

3) to conduct prior probing of the clouds using an aircraft or a small rocket when the cloud layer exceeds 1500 m in depth;

4) to have radio equipment at the launch site to determine a bearing for the "atmospherics" and to count thunderstorm discharges;

5) when clouds are present, to measure the current in a point discharge at the Earth's surface in order to detect the presence of a high intensity electrical field, which favors the formation of lightning.

In clouds of thickness 1500 m and more, there is a high probability for separation of electrical charges, which promote a storm discharge. These processes, as has been mentioned, are particularly dangerous, since it is much more difficult to observe them than storm clouds.

FLIGHT SAFETY OF MANNED SPACECRAFT /221

Under emergency conditions, the activity program of the crew should ensure that it is possible to cut short the flight and return safely to Earth. In order to make this possible, methods are being developed for evaluating an emergency situation as well as methods for rescuing the crew.

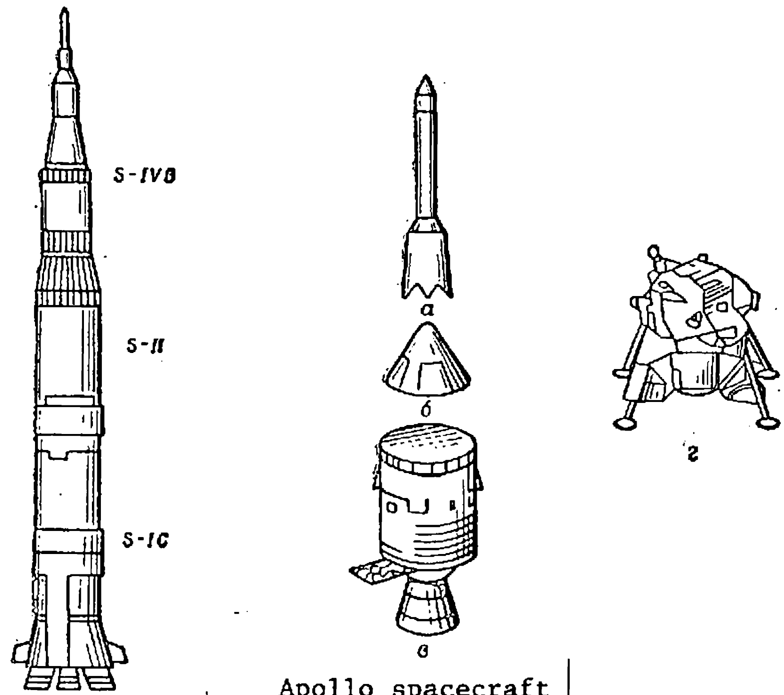
To evaluate emergency situations in the active part of the trajectory is a very difficult task, and depends on the ability of the ground services and the crew to keep track of the most critical systems and parameters. The method of crew rescue is very often determined by the nature of the deviation of the flight parameters from nominal values. In order to determine a program of action for a combination of all possible emergency situations, one must study each emergency situation, taking into account the effect of different limitations on the launch vehicle, the spacecraft, and the crew. These limitations can be due to both technical and operational factors. The program of action for emergency situations is described in several documents, each defining the degree of responsibility of the users. The period when the emergency situation arises affects the emergency action program. It is very important both for the ground flight control facilities, because of the need to adopt solutions and to protect the crew, and also for the crew members, who must carry out the required actions for emergency curtailment of the flight. By modeling emergency situations, the specialists have

developed a preliminary plan of action for an emergency, which is being checked and developed by discussion in working parties, by operations of the crew members on simulators, and by flight tests. The result is that launch vehicle and spacecraft hardware limitations will be determined and a plan of action during emergencies will be developed. It includes a list of emergency situations, instructions to the crew in case of emergency, a plan for search and rescue of a crew engaged in an emergency, a program of real time action for the emergency rescue system, and flight limitations.

We consider, in the lunar spacecraft example, methods of developing a program of action during emergency situations and a technique for crew rescue in different parts of a flight to the Moon, as developed in the USA.

Injection of spacecraft into orbit. Because of possible loss of launch vehicle thrust or explosion of the launch stage, this is one of the most hazardous sections of the flight. The atmospheric conditions are operational limitations, which require special examination of the technical parameters of the rescue equipment. Therefore, the spacecraft must be equipped with an emergency rescue system for the flight section within the atmosphere and under the launch conditions. Figure 82 shows the emergency rescue system, and the command, motor, and lunar modules, of the Apollo spacecraft.

The spacecraft design trajectory during atmospheric flight determines the initial conditions in the development of an emergency situation. The ground command point and the spacecraft crew must determine the deviation of the actual trajectory from the design one during the flight. The allowable deviation depends on the possibility of rescuing the crew at that given moment.



Saturn 5 launch vehicle

Apollo spacecraft

Figure 82. Components of the Saturn launch vehicle and the Apollo spacecraft.

a- emergency rescue system; b- command module; c- motor module; d- lunar module.

For the atmospheric flight section, we consider three basic types of emergency situation.

Loss of thrust of the launch vehicle motors. The most dangerous are emergency situations which arise in the initial flight section, prior to injection into orbit. The decision to make an emergency curtailment of the flight in this case is taken by the command point, from the data of the ground trajectory tracking equipment.

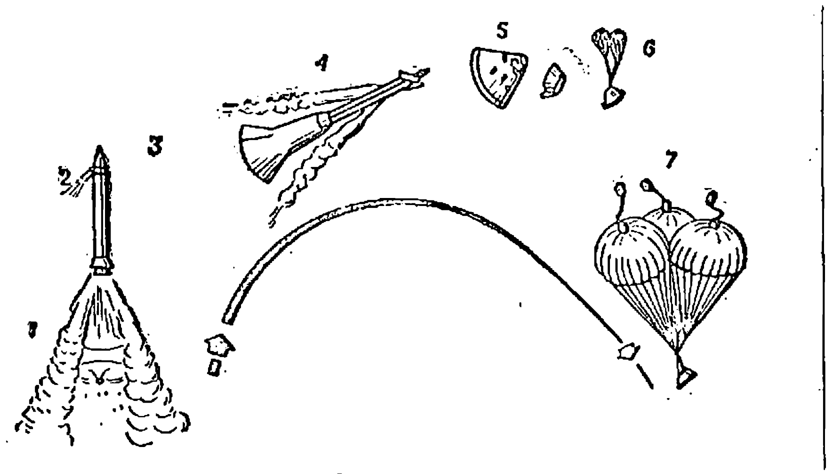


Figure 83. Operation of the emergency rescue system at low altitude.

1- ignition of the emergency motor; 2- ignition of the pitch motor; 3- deployment of the stabilizing fins (at 11 sec); 4- jettisoning of the truss (at 14 sec); 5- jettisoning of the cover (at 14.4 sec); 6- deployment release of the stabilizing parachutes (at 16 sec); 7- release of the main parachutes.

Rapidly developing emergency situation resulting from explosion or loss of stability of the launch vehicle. The nature of these emergencies, critical from the time point of view, require rapid appraisal of the crew situation, formulation of a decision, and initiation of the emergency rescue system without the aid of the ground service.

To evaluate the most critical flight situations in the Apollo spacecraft, there is an automatic system for determining the emergency environment. By measuring the angular velocities of the launch vehicle and the pressure in the motor combustion chambers, this system can initiate the emergency rescue system (Figure 83) in case these parameters exceed allowable values, which are angular velocities of 3 deg/sec in pitch and yaw, and 20 deg/sec in roll. At the same time as the emergency rescue system is initiated, the launch vehicle motors are cut, if the

emergency occurs 30 seconds after their ignition. During the first 30 seconds of the flight, the launch vehicle motors are not cut, to allow the launch vehicle to withdraw to a safe distance from the launch pad. The crew initiates the emergency rescue system from the readings of the angular velocity and angle parameters and from parameters which are tabulated on the spacecraft instrument panel. The automatic system for determining an emergency situation, combined with possible objective evaluation by the crew as to its nature (from the readings and physiological sensations) is a reasonable method for reducing the time to decide to cut short the flight for these emergency situations.

Slowly developing malfunctions whose effect on the launch vehicle flight is not evident to the crew (e.g., malfunction in the navigation and control system). The magnitude of slowly developing deviations from the design value is identified on the meters of the ground control center. The magnitude of allowable trajectory deviations are determined by the limits associated with the crew rescue system, particularly by the allowable loading during atmospheric entry following emergency separation. In the orbit injection section, four critical regimes can be identified: ignition of the launch vehicle, passing through sonic speed, flight at maximum dynamic head, and flight at high altitude. With an explosion of the launch vehicle at ignition, the emergency rescue system must rapidly remove the command module to a distance which is safe regarding the explosion wave and the high temperatures. Also, enough height and velocity must be provided for the deployment and filling of the parachute system. /224

In emergency separation at near sonic speed, additional difficulties arise, associated with aerodynamic interference between the command and motor modules. When the spacecraft separates under these conditions, an additional force is created due to the presence of an expansion in this region, in the area where the spacecraft unites with the engine module. This force acts at a distance of one diameter (approximately 3.6 m) of the spacecraft and specifies requirements as to minimum thrust capability of the emergency separation system.

Under conditions where the velocity head is maximum, a critical factor in emergency separation of the spacecraft is the shock action of the flame of the emergency separation motors on the structure. In the event of loss of stability of the launch vehicle, its angle of attack increases. During emergency separation under these conditions, the air stream deflects the motor flame and directs it onto the conical surface of the spacecraft, thereby creating critical loading conditions. In addition, with increase of flight height, the dimensions of the flame jet of the solid emergency motors increase, and it would entail an increase of mass to protect the spacecraft structure from the flame action. To avoid this during emergency separation, combinations of initial angle of attack, angular velocity, and aerodynamic stability characteristics of the spacecraft are chosen, for which this rotation during operation of the emergency motors would not be very severe. Up to a height of 21 km, the stability of the spacecraft, when joined to the emergency separation system, can be ensured by aerodynamic means. At greater altitudes, the aerodynamic features become ineffective, and the intensity of rotation of the separated part will depend on the eccentricity of the thrust vector of the emergency motors relative to the spacecraft center of gravity and the initial angular velocity of the launch vehicle at the moment of separation. However, the size of the dynamic head and the

damaging effect of the flame of the emergency motors are considerably reduced for a period of several minutes before jettisoning the tower with its emergency motors. The launch vehicle flies outside the atmosphere on a ballistic trajectory. /225
The main problem in emergency separation at this stage of flight is to ensure angular positions and angular speeds of the spacecraft which will eliminate large loading during the atmospheric entry.

Depending on the emergency situation and the flight conditions of the launch vehicle, the problem of spacecraft crew rescue in the orbit injection stage can be solved by various technical means. The emergency rescue system of the Apollo spacecraft is based on the principle of separating the spacecraft by means of the emergency motors mounted on a special tower. This method is used in all flight regimes up to the second launch vehicle stage. Under normal flight conditions, the tower and the emergency motors are ejected immediately following second stage ignition. Depending on the flight height at the moment the emergency occurs, the emergency rescue operation can proceed as follows. From the launch up to height 9 km, the rescue consists of the following steps:

1) the emergency rescue system is ignited;

2) the launch vehicle motor is cut (no earlier than 30 seconds after the launch);

3) the spacecraft separates from the motor module;

4) the emergency motors and the pitch motor are ignited; propellant is jettisoned from the attitude control and stabilization system motor (if the emergency occurs during the first 42 seconds of flight);

5) the stabilizing panels are deployed in the nose of the tower (11 seconds after separation of the spacecraft).

This sequence is retained also for flight heights from 9 to 30 km and from 30 km until the tower is jettisoned with the emergency motors. The difference is in the sequence of deploying the parachute system. For emergency spacecraft separation in the first stage, two stabilizing parachutes are deployed after 16 seconds. If the emergency occurs at a height up to 30 km, the deployment of the parachute system is delayed to height 7.2 km. In case of emergency at a height of more than 30 km, the crew must ensure a specific angular attitude of the spacecraft using the attitude control and stabilization system, after the emergency motors cease to operate, in order to avoid large loads during atmospheric re-entry. When an emergency situation arises outside the atmosphere, when the tower and its emergency motors have been jettisoned, the crew safety is ensured by other means. Figure 84 shows how the spacecraft separates from the launch vehicle at great height. Variant I corresponds to the case examined above, where the tower and emergency motors are used. In Variant II, the spacecraft separates from the launch vehicle, along with the motor unit, by use of the thrust of the latter. In Variant IV, as in Variant III, but with larger interval between the emergency separation and the maneuver, one is dealing with the case where the emergency occurs several seconds before injection into orbit. In this case, the spacecraft motors can /226
be used to transfer the separated part to an orbiting trajectory, and then there is no provision for a fast return to orbit.

A critical moment in the choice of Variant IV is to determine the energy capability of orbit injection of the spacecraft following malfunction of the launch vehicle. In this case, the control center, using the station tracking readings and theoretical limitations, make an appropriate recommendation, and in case

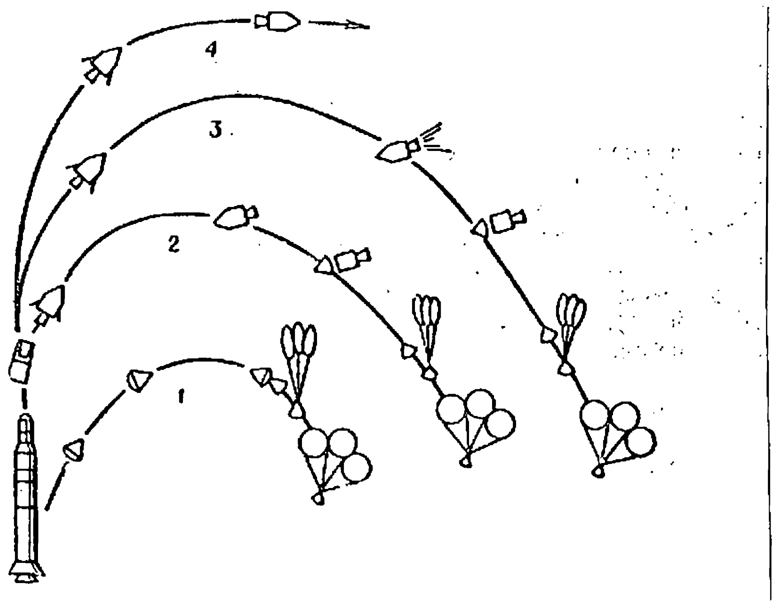


Figure 84. Variants of the emergency crew rescue procedures for the Apollo spacecraft during orbit injection.

1- Variant I; 2- Variant II (height up to 92.5 km); 3- Variant III (reserve); 4- Variant IV (speed greater than 7.3 km/sec).

of failure of communication with the crew, as indicated in the foreign literature, the crew uses special tables carried on board.

Transfer to lunar trajectory. This transfer is an active section in which the launch vehicle third stage creates the velocity increment required to overcome Earth gravity. It begins from an intermediate Earth orbit of height 185 km, and emergency situations do not force curtailment of the flight, as was true in the orbit injection stage, but can lead only to a change in flight program. For a program change, the allowable limits of deviation of flight conditions must be known. to retain the capability of emergency curtailment at any time. Since, in the Apollo spacecraft flight, the launch vehicle third stage is ignited above a given region of the Pacific Ocean, i.e., beyond

the limit of ground station tracking, the crew must evaluate emergency situations and take the required actions without the help of the control center. The causes of emergency curtailment of flight in the transfer to lunar trajectory section, as has been indicated, may be loss of stability and control and malfunction of the important life support systems. Failure of the third stage motor leads only to a change in flight program. To monitor the operation of the third stage, the crew uses tables showing the theoretical angular positions of the spacecraft and a program for the computers as a function of flight time during the burn of the third stage motor. These tables are designed both for automatic and for manual control.

To increase the crew safety, a technique was developed for accomplishing emergency Earth re-entry and cutting short the re-entry time, by simplifying the spacecraft maneuver and reducing the sensitivity of the emergency trajectory to scatter in the flight parameters. The emergency return operations can be accomplished without the help of the lunar control center in the injection to lunar trajectory section for the Apollo spacecraft. The spacecraft carries nomograms which the crew can use independently to perform a maneuver and determine the moment for ignition of the retro motor. In this way, a landing can be made in one of the five regions in which crew search facilities are located.

Flight to the Moon. The planning of emergency curtailment of free flight to the Moon, where there has been normal operation of the third stage and successful docking of the spacecraft with the lunar module, requires a choice of a motor to change the trajectory of the spacecraft, which consists of the linked command, engine, and lunar modules. The spacecraft is equipped with two independent motor units with self-contained stabilization systems for stabilization, navigation, and control. In addition, the spacecraft and the engine unit were equipped with independent

systems for attitude control and stabilization, which could be used to execute maneuvers. The spacecraft computer has a program for calculating trajectory data during emergency flight curtailment. This program could be used in the onboard computer. There was also a program which could be used to determine the required spacecraft velocity increment, depending on the time of occurrence of the emergency, and the required flight duration before return to Earth. In particular, the plan of emergency actions for the control center and crew was used during the flight of the Apollo 13, which experienced an emergency after 56 hours of flight. As a result of an explosion of an oxygen bottle in the engine module, an emergency situation arose in the command module, requiring rapid return of the crew to Earth. Up to the moment the decision was taken, the spacecraft remained in the lunar gravity field, and during flight in the emergency trajectory, a flight around the Moon was planned. During the spacecraft return, the crew used reserves of oxygen and water located aboard the lunar module, which was separated from the spacecraft only before atmospheric re-entry. Several plans of action had been developed /228
in general form for the flight section from launch to lunar approach, depending on the nature of the emergency situation.

Transfer to lunar orbit. A premature termination of operation of the main motor during transfer to lunar orbit required either emergency curtailment of the flight or a change in flight program. In the event of malfunction of the main motor, the energy capability of the lunar module must be sufficient to return to Earth. Because transfer to lunar orbit is accomplished on the back side of the Moon, the crew must evaluate this process without the help of the control center. During transfer to lunar orbit, an emergency situation can arise because of failure of the attitude control and stabilization system, failure of one or more systems not connected with the motor, failure of the motor and unpremeditated cutoff of the engine unit motor.

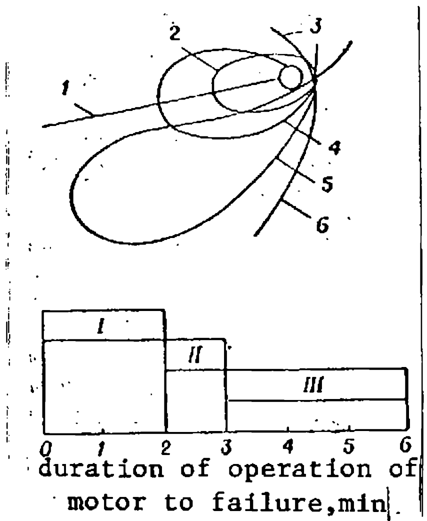


Figure 85. Possible Apollo spacecraft trajectories in event of failure of the main motor (depending on its duration of operation up to failure).

1- direction to Earth; 2- stable ellipse; 3- approach trajectory to the Moon; 4- ellipse intersecting the Moon; 5- unstable ellipse; 6- trajectory beyond the limit of lunar gravity.

For all these cases, recommendations were made up which use available instruments and graphs. When the main unit motor operates normally, the recommendation is that the crew complete the lunar orbit transfer. If the motor terminates prematurely, the following variations can arise (Figure 85) in the spacecraft trajectory: 1) return to Earth trajectory; 2) unstable ellipses and trajectories which involve collisions with the lunar surface; 3) stable elliptic trajectories.

In the first case, the spacecraft energy is sufficient to escape from lunar gravity. The second case occurs if there is engine malfunction in the first three minutes of operation. The third case results from engine failure at the end of its operation, when it is possible to change the flight program or to curtail it, and when an ordinary transfer from lunar orbit to Earth trajectory is performed. The maneuver can be done with the aid of the lunar module landing stage engine at the back side of the Moon.

Failure of the main unit motor during the first three minutes of operation leads to unstable trajectories and requires emergency curtailment of the spacecraft flight. In this case, the emergency source of thrust must be the lunar module motor, but difficulties /229
can be encountered in achieving spacecraft stability and control.

In event of failure of the main unit motor in the first three minutes of operation, which leads to unstable trajectories, the plan calls for the following operations. After preparing the landing stage motor and the LEM systems, a first correction burn of the motor is made. Then the spacecraft is oriented in such a way that the motor thrust direction coincides with the radius vector of the trajectory. Acquisition of a stable orbit allows the second burn to be made, taking into account the position of the Earth and the Moon, and the time factor is not critical. Then a transfer from lunar orbit to Earth trajectory is accomplished.

Transfer to an Earth trajectory. In the transfer section to Earth trajectory, the only source of thrust is the main unit motor, since the LEM has already separated. Maneuvers in this section are the same as in the transfer to lunar orbit section. In the event of premature cessation of the main unit motor in the transfer to Earth orbit section, situations arise similar to those resulting from engine failure in the transfer to lunar orbit section. However, the absence of the LEM increases the available energy capability of the spacecraft, since it reduces its total mass (in the event of premature curtailment of the motor), and in the absence of a duplicate motor unit, when there is complete failure of the main unit motor, return to Earth is impossible. For this flight section, the engineers have also developed emergency plans of action. Such emergency situations can also arise when the spacecraft (particularly one of Apollo type) is in lunar orbit, when the main unit motor fails in the transfer to Earth trajectory or if failure of the attitude control and stabilization system makes it impossible to return to Earth.

In all cases, the program of action in emergency situations in all stages of flight to the Moon and return reduces to choice of the best method of returning the spacecraft to Earth. This is accomplished while maintaining the capability of the spacecraft life support systems.

Thus, it is a complicated technical task, combining all aspects of space flight, to ensure safe flight of spacecraft.

ASTRONAUT TRAINING IN THE USA

In the period from 1959 to 1967, six sets of candidates for astronaut training were processed, and four of them included mainly pilots who had extensive experience in flight and had higher training. In the fifth and sixth sets, scientific workers were chosen with scientific degress in the philosophical, natural, and technical sciences. Later, in the seventh set, seven people were accepted in a seventh group. The result was that the group of astronauts numbered 73, of whom 53 remained at the end of 1969. Eight died (three in the Apollo spacecraft, four in aviation accidents, and one in an automobile crash). In 1971, 50 remained, of whom 16 were veterans who had already accomplished from one to four flights.

The theoretical concepts for the new candidates are presented during six months in a course of 568 hours. They study mainly flight dynamics and aerodynamics, control and navigation systems, physiology and life support systems, and also rocket motors, geology, and astronomy. The Moon flight candidates additionally study the lunar structure and undergo special preparation in astronomy aimed at a study of the position of 37 of the brightest stars in the equatorial region of the sky (these stars were used for Apollo navigation; data

on the stars is included in the computers). Then they have a course of study in a planetarium. The total duration of the course is 70 to 80 hours. At first, the astronauts practice identification of stars in the zodiacal constellations and determine the position of the Sun, Moon, and planets, and then they undergo training in order to establish stable permanent skill in recognizing the brightest constellations and determining their position relative to the zodiacal constellations. The final stage of study involves simulated flight in the equatorial region of the celestial sphere. In this period, the astronauts must know how to determine the latitude of their location relative to the celestial equator and measure deviation from the course stipulated in the flight program.

A study of space technology begins with the study of the Apollo spacecraft structure, to which 232 hours is devoted. Then the astronauts study individual spacecraft systems, and the launch vehicle. In order to fix their knowledge of the spacecraft, the astronauts take part in tests carried out in the workshops. In all, each astronaut spends 295 hours taking part in tests (monitoring and testing, on the launch site during preparation of the Apollo spacecraft).

On the launch site, the astronauts must take part in checking the docking of the main module with the lunar module, in tests of the emergency rescue system during spacecraft flight, and with the spacecraft on the launch pad, in spacecraft tests in an environmental chamber with simulation of external conditions during the flight, in pre-launch tests in preparation for launch, and in rehearsals of the pre-launch countdown.

Great importance is attached to the complex training simulators for the command and lunar modules, which are designed to simulate the spacecraft flight, beginning at the launch and ending with atmospheric entry. Each of these training devices consists of a mockup of the module cabin, equipment simulating the noise and the visual environment during the flight, and a control desk and a computer. In the cabin mockups, the internal equipment of actual cabins of the command and lunar modules is simulated. These are not sealed, but have equipment to create pressure in the spacesuits by supplying compressed air. In the cabin of the command module trainer, there is a system for removing life support products, containers with food and facilities for food preparation. The brightness of illumination in the cabin and the air temperature are controlled by the crew members. Magnetic tape recordings simulate the noise during operation of the launch vehicle engines, the main unit engines, the landing and engines of the main unit, landing and flight stage of the lunar module, the nozzles of the attitude control and stabilization system, as well as sounds during cabin decompression and noise in the radio communication system.

The optical system of the command module trainer has pictures in four windows and in the onboard telescope. Three forms of primary input are used in the optical system: star globes to imitate the sky, television units to present pictures of the lunar module after it has separated from the main unit, and projectors to show pictures of the Earth and lunar surfaces. The star globe is a large black sphere whose surface carries 997 steel balls of different sizes, corresponding to stars up to the seventh magnitude. The coordinates of the spheres on the globe surface are correct to an accuracy of five seconds of angle. The surface of some spheres are slightly colored, to simulate the color of the star light. The command module

trainer uses five star globes: one for the onboard telescope and four for the cabin windows. When the globe is illuminated, the light is reflected from the spheres and falls on the optical system, creating a picture of the star sky in the windows or in the telescope. The globe is mounted in such a way that it can be rotated and thus create a picture of only that part of the heavens which can be seen from the spacecraft windows at a given moment in the flight. Thus, the proper picture of the sky is created in front of each of the four cabin windows of the trainer and in the onboard telescope.

The television equipment for viewing pictures of the lunar module in the two forward cabin windows of the command module trainer consists of two cathode ray tubes. Each shows a model of the lunar module. When the model is moved in front of the two television cameras, the tubes show a picture of the lunar module in the form in which it could be observed under actual conditions in the right and left cabin windows. This equipment can be used to simulate docking of the command module to other spacecraft, by replacing the lunar module model by a model of a spacecraft.

The projectors are designed to show a model of the cabin module in the four windows and the telescope, and bright pictures of the lunar and Earth surface. Each projector has a rotary magazine containing photographs of different areas of the surface of the Earth and the Moon. On computer command, photographs are supplied to the projector corresponding to actual views of the Earth or Moon at the given moment of spacecraft flight. The optical system of the lunar module trainer is constructed similarly to the command module, and differs from it only as regards the equipment simulating a picture of the lunar surface during descent and landing of the module.

The equipment to simulate rendezvous maneuvers in rendezvous and docking of the lunar module and the main unit consists of two models of the main unit of different scale and a moving frame with two television cameras. The equipment is located in a dark room in the main unit: the model to simulate the rendezvous maneuver is at one end of the room, and the model to simulate docking at the other. These models are illuminated. By altering the frame and the television cameras, one can simulate approach to the main unit from 160 m up to the moment of docking.

The lunar trainer optical system can simulate landing and takeoff of the lunar module. The simulator depicts landing of the lunar module from a height of 366 to 1.2 m above the lunar surface. The main elements of this simulator are: a model of part of the lunar surface on which the astronauts must make the landing and a television camera. The increase in picture scale of the lunar surface features is shown in accordance with a programmed lunar module descent, and the unit also shows the turn of the lunar module from the horizontal position to the vertical, hovering of the lunar module near the lunar surface, and firing of the retro motor. During descent on the trainer into a crater, the astronauts can see the curvature of its walls. In particular, the model of the landing region of the Apollo 11 spacecraft has a mass of 270 kilograms and was made of epoxy resin, and showed all the details of the lunar surface, including features of size down to 0.9 m.

Control desks were provided for observing the actions of the astronauts while in the cabin models and their physiological indications, for monitoring the operation of the apparatus, and for introducing irregularities into the model in order to practice crew response in emergency conditions, and to develop habits in making contact with the spaceflight control center.

These desks were used to set up the required operating conditions in the trainers, to shorten the training period, and to return to various stages of the program for repeat exercises, i.e., the trainers were used to practice all activities necessary to carry out the actual flight program. The trainer computer unit received signals from the spacecraft panels and from the spaceflight control center. In response to the input signals, the computer developed control signals for equipment simulating the noise and visual effects and changed the readings of the spacecraft meters. While rehearsing the flight in the command and lunar module trainers in conjunction with the flight control center, the launch site computer and the spaceflight control center were switched into the trainer computers. /233

The preflight training of the astronauts continued for three months. In the last twenty days of this period, the launch site trainers were connected via ground communication lines with the spaceflight control center, thereby allowing monitoring of the astronauts' activities and of operation of equipment in the models of the command and the lunar module cabins, not only from the trainer control desk, but also from the flight control center.

It is mentioned in the foreign press that existing trainers have significant shortcomings: they do not show dynamic loadings and the state of weightlessness in which the astronauts exist during an actual flight. Therefore, there are special trainers to represent special stages of the flight program in conditions which simulate those of actual flight to the maximum extent.

There is an interesting dynamic lunar module trainer designed to simulate docking of the lunar module with the command module when they are distant 30 m apart. The trainer simulates limited motion in six degrees of freedom. There are also

trainers for practicing lunar landing and takeoff. The training for landing on the Moon in the final stage of preparation is conducted on a one-dimensional aircraft powered by a turbofan engine with a thrust of 1900 kgf and two rocket motors with a thrust of 227 kgf each, operating on hydrogen peroxide. The turbofan engine balances 5/6 of the weight of the aircraft, thereby simulating conditions of lunar gravity.

Much attention has been given to training the astronauts in the conditions of weightlessness and gravity existing on the lunar surface. For short periods of time, the state of weightlessness is created in parabolic flight of a KS-135 aircraft, equipped with a weightlessness "pool" which contains a model of the command and lunar spacecraft modules. The total duration in the weightless state in aircraft flights on a parabolic trajectory constitutes about 13 hours for each Apollo astronaut.

Long periods of weightlessness and lunar gravity are simulated in a water tank by creating neutral buoyancy or buoyancy which removes 5/6 of the astronaut's weight. The tank is of diameter 9 m and of depth 5 m and has metal models of the command and lunar /234 modules for practicing transfer of the astronauts from module to module in space conditions. For training of the Apollo astronauts there is an equipment simulating lunar gravity in which 5/6 of the astronaut's weight is removed by means of a harness system. The same trainer is used to practice movement on the Moon. There are other trainers for practicing the exit from the command module after normal splashdown in the ocean and when in the upside-down position, i.e., with the vertex of the cone in the water, which requires the use of floats to invert the command module.

The fire which occurred on January 27, 1967, during ground tests of the Apollo command module, led to the death of three astronauts and, therefore, a great deal of attention is now given

to practicing actions of the crew in the case of fire in the spacecraft. The crew practiced exit from the spacecraft on the launch pad, with emergency blowoff of the hatch and the use of onboard fire extinguishers during fire in the command and lunar modules.

The duration of individual periods in the trainers is not prescribed in the course. It is required only that the total time spent in training should be not less than 200 hours and the total time spent in the classroom should be not less than 60 hours. Usually, a session consists of instruction for an hour and a three-hour practice period. The special training of the Apollo astronauts, in which they studied the spacecraft and launch vehicle hardware and spent time on the various trainers, was designed to last 12 months (more than 2000 hours). In fact, it lasted from 6.5 to 14 months, as reported in the press, depending on the individual capabilities of the astronaut. During this time the astronauts spent 10 — 15 hours daily, and had one day off per week.

THE SKYLAB ORBITING LABORATORY

In the period after the first manned flight into space, astronautics achieved a great deal of success. During this small time interval, man's capability to live and work in space conditions was evaluated, men were landed on the Moon and returned to Earth, flights were made into space, and a space walk was accomplished. Rendezvous, docking, and transfer from one spacecraft to another were shown to be possible, and not one of these flights showed evidence of any appreciable psychological or physiological limitations as regards increasing the period that a human stayed in space. Subsequent flights will allow these limits to be determined, if they exist. /235

However, it should be noted that until recently, advances in space activity have been limited by the cost of putting a payload into orbit and by the capabilities of the hardware which was injected into space. Therefore, the next step is to develop economical rocket and space systems, comprising an orbiting station and multiple-use hardware for the Earth-space-Earth journey.

This equipment is a prototype of later space stations, which, like the first laboratory, will be used to conduct diverse scientific, engineering, biological, and other experiments. These stations are planned for use as orbiting laboratories to investigate and understand the nature of space, as extra-atmospheric observatories to observe the Sun, planets, and stars, as a base for continuous observation of the Earth and its atmosphere, for the resolution of meteorological and oceanographic problems, the discovery and evaluation of Earth resources, communication and radiotelemetry, monitoring the motion movement of typhoons, and for advance warnings of natural disasters. Such stations can be used for refueling, equipping remote mission spacecraft, and as intermediate bases for missions to distant orbits, to the Moon, and the other planets. The six-month flight of the Soviet Salyut laboratory marked the beginning of a new phase in the study of the Earth. Various objects and areas on the Earth's surface reflect a different amount of heat and light, which is picked up by photographic cameras and spectrometers. Cartographers, oceanographers, and geologists are interested, to different degrees, in studying the Earth's resources from space. It has been determined that man presently uses only 1 in 3200 of daily energy which the Earth receives from the Sun. Therefore, it is very important to improve our understanding of the Sun, its activity, and its effect on the Earth. There are questions associated with space astronomy, to be studied and resolved, which can only be examined by orbiting laboratories.

The possibility of developing production processes in space is one of special interest. The reason is the unique absence of gravity existing in orbit. We can expect that the weightlessness state will allow high quality foam materials to be made from any mixture of gases and liquids. Many technical processes previously considered impossible may be possible in space.

The technical processes which may be accomplished onboard an orbiting laboratory include the following:

1. Production of ball bearings with very low deviations from perfect shape. For this purpose, the metal must be located in a vacuum chamber where it will be melted as a result of high frequency heating. Under the influence of surface tension, the /236
molten metal acquires a spherical shape. The time required to produce spheres will depend on the time necessary to "kill" the molten metal. Deviations in the shape of spheres from perfection will be at least three orders of magnitude less than for terrestial conditions, in the opinion of the specialists.

2. Production of articles of different configurations. For example, an alloy, exposed to the action of a magnetic field, can be drawn along only along one axis, and takes the shape of an ellipse. By creating a magnetic field controlled by a computer, one could produce articles of very complicated shape. If the fused metal were rotated, it might take a disc shape, and if the rate of spin were sufficiently high, the shape of a torus.

3. The production of foam-metal. In terrestrial conditions, this process is impossible because of the very low viscosity of the molten mass. One possible method for producing foam-metal is to introduce the metal and the gas simultaneously in a vacuum chamber. Another method is to use a mixer to agitate the molten metal while the gas is injected, and if the rate of mixing is

high enough, bubbles of gas in the metal can coalesce and form a torus. A third method involves injecting the gas into the molten metal at high pressure, and thereafter, instantly inserting the metal into a vacuum chamber. If the pressure drop is sufficiently rapid, the gas bubbles distribute themselves uniformly in the foam-metal.

4. The production of filament-shaped monocrystals, particularly monocrystals of sapphire and beryllium oxide.

5. The production of alloy layers by a method in which a layer of metal with high melting temperature is poured onto a layer of metal with low melting point. In this case, there is not mixing of the layers, or almost none, because of their surface tension and their inherent tendency not to mix.

6. The production of high temperature alloys for which there are no containers on Earth capable of standing their melting temperatures. Under weightless conditions such alloys can be obtained without the use of containers by high frequency heating in a magnetic field.

7. The production of new kinds of glass from oxides of titanium, zirconium, and hafnium. This kind of gas cannot be obtained in terrestrial conditions, because there are no containers capable of withstanding the melting temperatures of these oxides.

The possibility is also considered of using direct solar energy for certain technological processes. With this objective it is proposed to use a parabolic reflector of diameter 100 m, with 100% reflectance, which could give a power of 11 MW. This would be enough to melt 2.3 kg of copper per second.

A good deal of attention has been given in the USA to creating an orbiting laboratory. The Americans planned to use such a laboratory to carry out a program of scientific investigations, including astronomical, technological, medical, and biological, and other experiments. This kind of laboratory was based on elements of the Apollo spacecraft and the Saturn launch vehicle. The Skylab station (Figure 86) consists of the following main parts:

— modified main unit of the Apollo spacecraft (command and engine modules), designed to put a crew onboard the laboratory;

— a general purpose mooring structure for docking with the other laboratory units and to hold a control panel, scientific instruments, and stocks of photographic film; this unit would also include power gyroscopes for stabilizing the laboratory;

— a lock chamber joining the mooring structure to the laboratory module, and for transferring astronauts from the main Apollo unit to the laboratory;

— an instrument module for altitude control of the laboratory, using nozzles operating with compressed nitrogen, until the power gyroscopes of the special unit would come into action;

— a gyroscope unit with a group of astronomical instruments equipped with two panels of astronomical solar cells; this unit would also contain the power gyroscopes for altitude control of the laboratory; /238

— the orbiting laboratory which is a modified stage of the Saturn launch vehicle, to function as a living quarters module and a laboratory; the laboratory has its own panels of solar

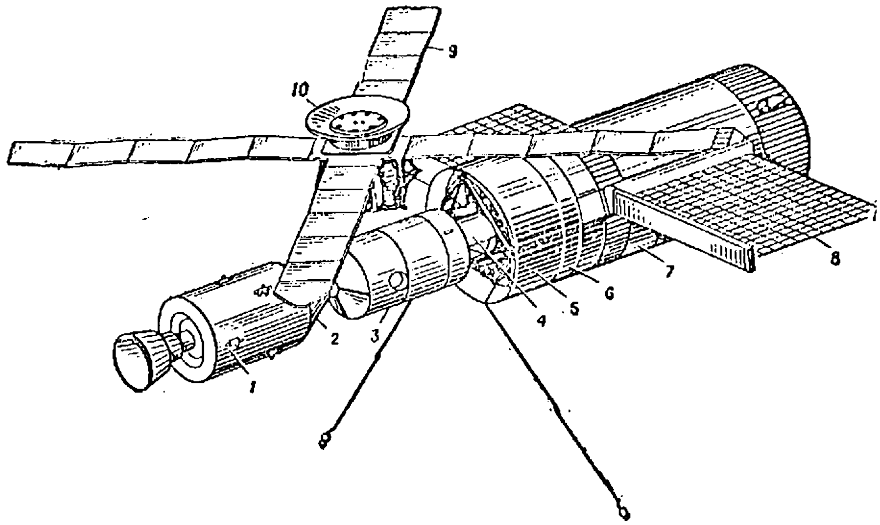


Figure 86. General view of the Skylab laboratory:

1- service module; 2- command module; 3- universal docking transfer adapter; 4- lock chamber; 5- lower payload fairing; 6- instrument module; 7- Saturn IV B stage (orbiting laboratory); 8- solar cell panels of the SIVB stage; 9- solar cell panels of the ATM (power gyroscope) unit; 10- bank of astronomical instruments of the ATM unit.

cells and attitude control system, powered by jet nozzles; the modification of the launch vehicle stage involves installing some of the equipment of the laboratory inside the hydrogen tank of the stage, and welding the stage to the lock chamber and the mooring structure.

The general purpose transfer unit is a geometric cylinder of length 5.2 m and diameter 3 m (Figure 87). The transfer unit has two docking fixtures (one is a reserve). On opposite sides of the cylinder walls there are two windows: one to observe during experiments using the gyroscope unit telescopes, and the other for observation of the Earth. Inside the transfer unit there is some of the equipment for performing the experiments, as well as the control desk for the gyroscope unit instruments.

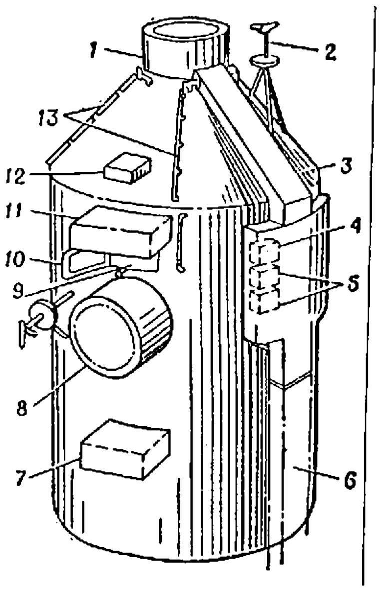


Figure 87. Universal docking adapter:

1- end docking fixture (along the axis); 2- docking attitude fixture; 3- tunnel for electrical gear and instruments; 4- signal system; 5- electrical distributor distribution; 6- tunnel communicating with the main laboratory module; 7- multi-spectral scanning unit; 8- side dokcing fixture (reserve); 9- window; 10- window flap; 11- IR spectrometer; 12- spectrometer for proton investigation; 13- handrails.

Scientists are also studying the possibility of using the general purpose docking and transfer unit as a shelter for the astronauts during intense solar flares or during an emergency of the major laboratory systems, when the astronauts must await help from Earth.

The lock chamber, of length 5.1 m and diameter 1.65 m, consists of a room containing supplies of gases to create the artificial atmosphere, an electrical distribution and communication systems, and the tunnel. The lock chamber has a hatch for the astronauts to go outside into space (it allows two astronauts to exit into space without resealing the adjoining rooms of the laboratory).

The gyroscope unit is designed for the conduct of astronomical investigations, and is an eight-sided truss within which there is a cylindrical container of length 3.4 m and diameter of about 2 m. A cross-shaped partition divides the internal volume of the cylinder into four parts and serves as a support for attachment of the telescopes. A cooling liquid circulates to cool the cylinder and its lining, so that the temperature of the inner surface of the cylinder does not exceed 10° C. The mass of the /239

gyroscope unit is about 10 tons. The unit contains the telescopes and five television cameras, with which the astronauts can observe during the experiments. Control of the operation of the instruments mounted in the gyroscope unit is carried out remotely from the general purpose docking and transfer unit. Stabilization of the gyroscope unit is accomplished by means of three power gyroscopes. The accuracy of attitude control is 2.5 seconds of angle.

The service system for the Skylab includes an electrical system, a system to control the ambient conditions, and an attitude control system.

The laboratory electrical system consists of solar cell panels, conversion systems for conversion and distribution of electrical power, illumination and communication systems, and an emergency signal system. The batteries in the gyroscope unit and the lock chamber are used when the laboratory is in the section of its orbit not illuminated by the Sun. The electrical power is wired to the lamps, the communication system, the experimental equipment, the emergency signal system, the life support system, the system for gasdynamic attitude control, the telemetry system and the sensors. In addition to a large number of sockets and plug joints, designed for use with portable equipment (television cameras, fans, auxiliary lamps, and a vacuum cleaner), the lighting system also includes the general and emergency subsystems. Some of the general lamps of the general lighting system are used when the crew return to the living quarters inside the laboratory, and these have a self-contained supply. The additional level of lighting for photography and other purposes is provided by means of special portable lamps.

External lighting is provided (in the lock chamber and the docking transfer unit) for operation in external space. There are also signal flares used for warning purposes when the main Apollo unit approaches the laboratory. The emergency illumination system is switched on automatically if the voltage drops in the mains.

The communication system provides voice communication between the crew and the Earth. Internal communication telephones are provided in the modules of the laboratory. The internal communication system has an outlet at a transmitter located in the main Apollo spacecraft unit. The laboratory has a teleprinter unit connected directly to the command transmission system. This unit makes copies of communications received from Earth. To determine the range when the Apollo spacecraft makes a rendezvous with the laboratory there is a device whose units are located in the Apollo command module and in the lock chamber. The maximum measurable range is 526 km. There are emergency signal sensors in all the laboratory modules. Signals from the sensors, which /240
are sensitive to a sharp drop in pressure and to increase in temperature (in case of fire), go to an emergency panel in the lock chamber. Simultaneously an alarm is sounded in all the speakers. The emergency signals must also be transmitted to Earth over the telemetry.

The system for monitoring external conditions consists of subsystems for pressure regulation, temperature regulation, fans, and atmospheric composition regulation. Before launch, all the laboratory modules are filled with nitrogen, which gradually leaks away during injection of the laboratory into orbit. The pressure control subsystem, located in the lock chamber, controls the total pressure in the inside rooms of the laboratory, as well as the partial pressure of oxygen. When the crew leaves the laboratory, the internal pressure drops from 0.35 to 0.035 atm,

by natural leakage. A pressure level of 0.035 atm is maintained by command from Earth during the holding period. When a replacement crew returns to the laboratory, the command from Earth is given to raise the pressure again to 0.35 atm. Ventilation of the atmosphere, control of temperature, and monitoring the elimination of harmful impurities is performed by a central subsystem located in the lock chamber, and also by means of heat transfer fans, thermal regulation coatings, and filters in the various modules of the laboratory.

Skylab has two attitude control systems. One is used to orient the laboratory following injection into orbit, and stage separation. The control moments at this time are generated by a gasdynamic system, operating with compressed nitrogen, and using six nozzles located in the stern of the instrument module. The main attitude control system, which uses power gyroscopes (inertial flywheels) is located in the power gyroscope unit.

Signals from the sensors which monitor the direction to the Sun, in the star orientation and rate gyros, go to the onboard computer in the power gyroscope unit, where the spacecraft attitude is calculated and command signals are generated for the power gyroscopes.

The attitude control system can operate in several regimes, chosen by command from Earth or from on board the laboratory. The regime can be heliocentric orientation, rotation of the laboratory towards the main Apollo spacecraft unit for rendezvous and docking, orientation along the local vertical for observation of the Earth, and with the attitude fixed in an arbitrary direction.

The Skylab was injected into orbit at a height of 435 km, and an inclination of 50°, by means of the Saturn 5 launch vehicle. After orbit injection and separation from the launch vehicle second-stage, the laboratory was turned through 180° (Figure 88), during which time the forward fairing was jettisoned. Then the power gyroscope unit was made ready, and the entire laboratory was rotated with respect to the vectorial direction to the Sun. The re-orientation process took roughly the time of one-quarter revolution, and then the solar cell panels and the meteorite shield screen were deployed, and the rotors of the power gyroscopes were also brought up to speed. /241

A constant heliocentric attitude (the axis parallel to the Earth-Sun vector) is maintained during the whole period that the laboratory is operating, apart from periods of rendezvous and docking with the Apollo spacecraft, and during experiments to search for Earth resources.

The launch of the second rocket in the Skylab program was carried out one day following the first launch. Its task was to use a two-stage Saturn lanch vehicle to inject into orbit the Apollo main spacecraft unit, with a crew of three. After rendezvous and docking of the main unit and the laboratory, the crew transferred from the spacecraft command module to the laboratory module, and then the main unit systems were switched off and parked. The first crew was to stay aboard the laboratory for 28 days. Their main mission was to conduct medical investi- /242
gations, to assess the suitability of the internal rooms of the laboratory and its equipment for a long residence of humans in space, as well as to check and unpack the scientific instruments of the power gyroscope unit.

The assignment of the second crew will be to continue these medical and biological investigations in order to improve one's

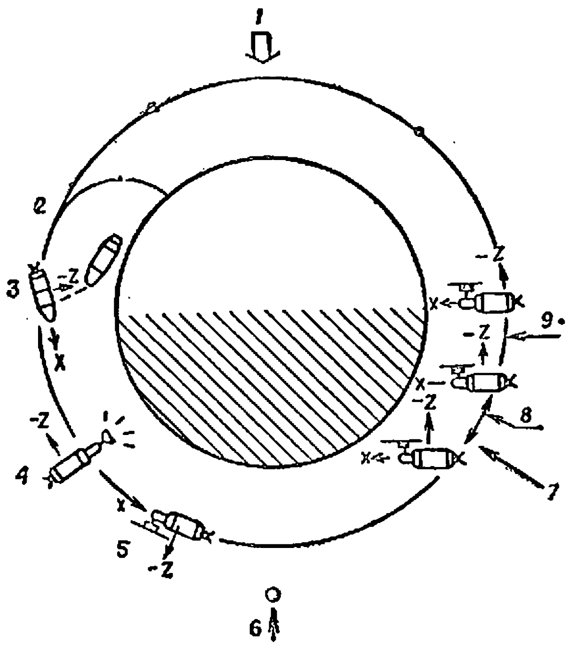


Figure 88. Sequence of operations in preparing the Skylab laboratory:

1- Sun vector; 2- stage adjustment of S-II stage attitude at the orbit injection point; 3- separation of S-II stage and rotation of payload through 180°; 4- jettisoning of forward main fairing and beginning of deployment of ATM unit; 5- deployment of ATM solar cell panels, ATM unit being prepared; 6- midpoint of shadow section of orbit; 7- end of re-orientation of Skylab in helio-stationary position; 8- deployment of solar cell panels of orbiting laboratory; 9- beginning of despinning of gyroscope power rotors (laboratory ready for work).

knowledge of deep space conditions on humans. Therefore, the second crew will include a doctor. The proposed period of this crew onboard the laboratory is 56 days.

The third crew is to carry out a program of scientific, engineering, medical, and biological experiments, also for a period of 56 days.

The sequence of laboratory launches is shown in Figure 89. The main module of the orbiting laboratory (modified stage of the Saturn 5 launch vehicle; Figure 90) is intended to carry a crew of three astronauts (like the Soviet Salyut laboratory). The inside of the module is divided into a living quarters and laboratory modules by means of aluminum screens. The living module in turn is divided into the following rooms: for sleep (three sleeping bags are provided); for recreation time, and preparation and consumption of food; for personal hygiene purposes; and for training and conduct of experiments.

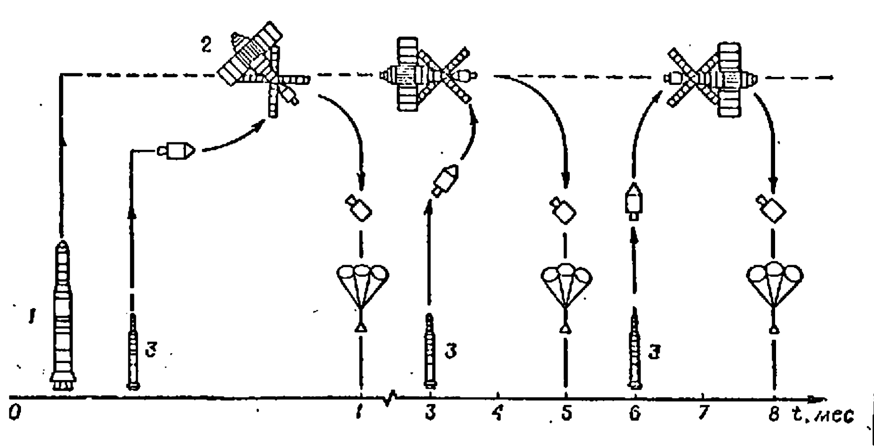


Figure 89. Sequence of launches in the Skylab program:
1- orbit injection of main module (modified SIVB stage) without crew; 2- remote switching on of systems and preparation for work; 3- main Apollo spacecraft unit with crew.

In order to develop the living quarters module for the Skylab, it was proposed that a 56-day experiment be conducted under terrestrial conditions, simulating space conditions. The object of the experiment was a medical and biological investigation and an appraisal of the medical equipment to be carried on the laboratory. A chamber of diameter 6 m, designed for this experiment, was used earlier for experiments in the Gemini and Apollo programs. The chamber was filled with a two-gas atmosphere (70% oxygen and 30% nitrogen) at a pressure of 0.352 kg/m^2, and a partial pressure of carbon dioxide of not more than 4.0 — 5.5 torr. The chamber was maintained at a temperature of 20 — 25 C, and a humidity of 45 — 60%. The test required the same food and water as for the members of the Skylab crew. A closed circuit television system was used to observe the progress of medical and biological investigations. The experiment in the chamber required the participation of the three subjects who conducted the medical and biological investigations planned for the orbiting laboratory. In the laboratory, 16 investigations of this kind were carried out, including investigations of the /243

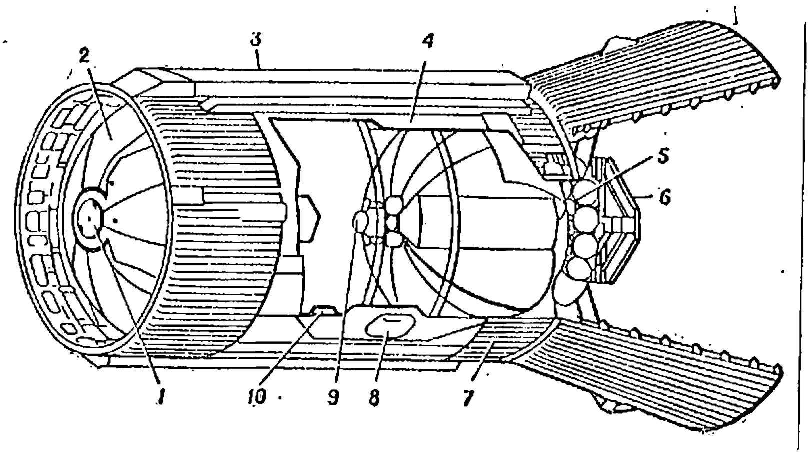


Figure 90. Main laboratory module:

1- hatch for access to lock chamber; 2- high quality insulation; 3- solar cell panels; 4- meteorite shield; 5- jet system for attitude control; 6- radiator; 7- heat shield; 8- access hatch for servicing on Earth; 9- waste disposal port; 10- lock chamber for withdrawal of scientific instruments into space.

effect of space flight conditions on the cardiovascular system, the energy expenditure during physical stress, and reactions of the digestive system.

Besides the medical and biological investigations, the subjects operated experimental equipment. The subjects were under constant medical supervision. The living area was separated into three individual bedrooms, a bathroom, and a common room. The astronauts slept in sleeping bags attached to the wall. The sleeping position was unusual, perpendicular to the floor of the living area; under conditions of weightlessness this has no significance, but in a structural sense, these sleeping positions were easier to arrange than horizontal ones, according to the American designers. Each sleeping berth had a locker for night articles and a receptacle for waste. The bathroom equipment was: a toilet for biological waste, and a /244
freezer to store specimens of this waste, designed for laboratory

analysis on Earth, a vacuum cleaner, a wash basin, a waste receptacle, and a cabinet for storing toilet articles. The common room of the living space was used for cooking, dining, and study. The living quarters had a refrigerator and food cupboards, recording devices, three stoves and a table for food preparation, a faucet for drinking water, cabinets for napkins and kitchen towels, and also waste receptacles.

In preparing the launch of the orbiting laboratory, a full scale model of the main laboratory module was constructed, a modified Saturn rocket stage. It had a length of 15 m, a diameter of 6.5 m, and a mass of 27.5 tons, and was fully equipped internally, just like the flying model.

Then the Skylab shroud, which had a length of 17 m, a diameter of 6.7 m, and a mass of 11 tons was tested in a space chamber. It consisted of four sections of panels, which were required to fly off without touching one another, with a speed of about 20 km/hr, during tests of the blowoff mechanism. The purpose of the tests was to check the operation of the blowoff mechanism, which must ensure correct removal of the panels inside the space chamber, simulating conditions in space at a height of 580 km.

On May 14, 1973, the USA launched the Skylab laboratory, but the thermal protection screen was damaged during injection into orbit and did not uncover one of the panels of solar cells.

Launch of the crew, consisting of Weiss, Cerwin, and Conrad, for the Skylab, planned for May 15, was rescheduled for May 20, 1973, because of a fault in the laboratory electrical supply. Then the launch was again rescheduled for May 25, because of the need to explore the possibility of deploying a new thermal

protection screen to maintain normal temperature inside the laboratory, which would otherwise get as high as 53° C, because of the damage to the main screen during injection into orbit.

The mass of the laboratory injected into orbit was 77 tons, and its mass, when docked with the Apollo spacecraft, was 90 tons. The length of the laboratory and the spacecraft was about 35 m, and the maximum diameter was 6.6 m. The volume of the rooms with artificial atmosphere was more than 300 m^3.

First, the astronauts mounted the thermal protection screen on the laboratory, and then they could enter the interior. Before this, they stayed in the Apollo spacecraft cabin, in which there were screens taken from Earth, and other equipment required to repair the laboratory. After crossing over to the laboratory, the astronauts had to repair the solar cell panels, one of which had not deployed, since a fragment of aluminum had fallen into it wedging the panel. An attempt to remove the fragment by flying the spacecraft around the laboratory had not succeeded (Weiss had reached out from the hatch, and Cervin had held his legs so that he would not fall out). At the same time, Conrad controlled the Apollo spacecraft. /245

Because of an acute shortage of electrical energy aboard the laboratory, it was decided on May 30 to try again to unfold the undeployed solar cell panel. To do this, it was necessary for one or two astronauts to make an excursion out into space. The temperature inside the laboratory was 26 — 28°, which presented additional difficulty in performing the necessary experiments to be done by the astronauts. On June 4, after consultation with Earth, it was decided to make an excursion outside on June 7. Conrad was to do this. The space excursion would last 1.5 — 2 hours. At this time, the astronaut had to go approximately 6 m from the hatch in the docking lock chamber

to the undeployed panel, and this was done. Conrad removed the bolt from the anti-meteorite screen, which had been broken in the first minutes of the laboratory flight, using a scissors mechanism, which the two astronauts operated, one with the help of a seven meter cable, attached to one link of the scissors. As soon as the bolt was removed and the panel was free, the astronauts used the cable to release the panel from the spacecraft body. All three sections of solar cell panels were opened, and charging of the eight batteries which were connected to the undeployed panel initially was begun. In the deployed condition, the panel measured 8 x 4.9 m, and its mass was 912 kg.

Because conditions were now normal, it was decided that the crew should make a flight of 28 days, as scheduled in the program. However, immediately after they performed their space excursion, a valve in the main laboratory main cooling system malfunctioned, and the astronauts had to switch to the spare system.

During the Skylab flight, they performed medical experiments, astronomical observations, and investigations of natural resources. The technical operations under conditions of weightlessness and vacuum (similar to those conducted earlier by Soviet astronauts) were very interesting. The technical operations in the American spacecraft were carried out in a special spherical chamber of diameter 40 cm, located in the mooring structure.

The chamber opened up to the outside space to create a vacuum and had a window for visual observation, photography, and television photography of the process. The chamber was equipped with an electron beam welder, an electrical heater, and an oven for crystal growing. The first experiment to be done was welding of metals using the electron beam, which heated the weld to temperatures above 1100° C. Welds were done with strips of stainless steel, aluminum, and nickel alloy. Then

they made spherical castings with the nickel alloy (two spheres of diameter about 6 mm) to study the container-free melting and solidification of the specimens. /246

On June 16, after the program was completed, they carried out a three-hour training period in the Apollo spacecraft cabin, while docked with the laboratory. The purpose of this training was to develop operations for undocking, departure from orbit, and return to Earth. On June 18, after a flight of duration 28 days and 48 minutes, the astronauts parked the onboard systems and prepared to leave the laboratory. On June 19, Air Force Major General Shatalov, twice a hero of the Soviet Union, congratulated the Skylab crew on their achievements and wished them a successful return to Earth. On the same day the American astronauts made an excursion into space to accomplish certain auxiliary operations. They were in good spirits, and joked a good deal. During this work in space, the two astronauts were tethered by halyards of length up to 18 m. Conrad changed the film cassettes, and removed contamination from the objective lens of the coronograph mounted on the spacecraft. On June 22, the spacecraft was undocked from the laboratory, the longitudinal engine of the Apollo spacecraft was ignited, and the spacecraft transferred to a lower orbit. Then they initiated the retro burn and descended from orbit. At the end of the entry, the braking parachutes opened at the scheduled height, and the crew module splashed down at a distance of about 30 km from the aircraft carrier awaiting them. The astronauts' capsule was hoisted to the deck of the carrier. The Skylab crew took about 20,000 photographs of the Earth and 30,000 photographs of the Sun and other astronomical objects.

On July 28, 1973, a second crew, Bean, Loomis, and Herriot, were sent up to Skylab in near Earth orbit. Previously Bean had taken part in the Apollo 12 missions and had landed on the Moon.

On September 26, 1973, after a duration of 59-1/2 days, the flight of the second Skylab crew came to an end. At the start of the flight, the astronauts had to set up a supplementary thermal protection screen to lower the temperature inside the laboratory (the temperature had increased because the "umbrella" set up above the laboratory by the first crew had not opened completely). In particular, it did not cover the part of the laboratory containing the water tanks, whose temperature had risen to 49° C. Thereafter, the astronauts proceeded with their main mission, investigation of the Sun using the set of astronomical instruments, and observations undisturbed by the Earth's atmosphere. A feature of these investigations was observation of a solar flare which liberated energy equivalent to the explosive energy of 100,000,000 million tons of TNT.

Some interesting biological investigations were conducted by the second crew. For example, two spiders taken aboard in the first days of the flight lost their orientation in the weightless condition and spun their webs in a random manner in the corners of the glass container, and then later began to create a geometrically correct pattern, as on Earth. Fish, also carried aboard, differed from the spiders in that they did not become adapted to conditions of weightlessness, but continued to swim jerkily in a spiral fashion. However, their offspring, hatched from eggs, immediately began to swim in a straight line. Therefore, it was suggested that adaptation to the weightless condition apparently took place in the embryonic state. /247

One of the mission tasks was to investigate natural resources and also to study environmental impurities.

Tests were conducted to obtain alloys in the onboard electrical furnace. Metals were fused in special capsules under conditions of weightlessness, metals whose alloys cannot be

obtained on Earth, since separation of components occurs under the influence of gravity forces. During another experiment in the electrical furnace, melting and evaporation of two crystals was done, the objective being to study the possibility of increasing the quality and the chemical homogeneity of crystals obtained from the vapor phase in weightless conditions. If it is possible in weightless conditions to obtain crystals containing less impurity than crystals prepared under gravity conditions, then industry will have new possibilities for preparing super-strong materials, as well as qualitatively more perfect crystal components, especially for electronics. This would lead to a considerable reduction in the price and an improvement in the quality of laser equipment, television equipment, and computers.

The landing of the Apollo transport spacecraft, on which the astronauts transferred from the Skylab, was carried out under difficult conditions, since only two of the four auxiliary spacecraft engines were operational. Therefore, a new procedure for control of Apollo had to be developed in the ground trainer.

On November 16, 1973, an Apollo spacecraft was launched, carrying the third crew to Skylab.

According to the press reports, during the descent to Earth, the astronauts of the third crew were forced to use an emergency landing system, since the main system for activating the spacecraft retro motors was unserviceable. The astronauts Carr, Gibson, and Boyd lived through 45 critical minutes, to use their words, when they were in the zone above the Earth when there is no communication between the spacecraft and Earth. They observed, to their horror, that the spacecraft retro motors had not ignited.

It was not possible to obtain instructions at this moment. The spacecraft was in danger of entering the atmosphere at the incorrect angle, which could have serious consequences, both for the spacecraft and for the astronauts. The astronauts accomplished the landing manually, using the emergency system.

CHAPTER 8

RE-USABLE SPACECRAFT. FLIGHTS TO OTHER PLANETS /248

The descent of a spacecraft to Earth, as we have noted, is carried out using aerodynamic control of the flight.

One of the important characteristics of any entry vehicle is the available (maximum) aerodynamic lift, which determines the possibility of aerodynamic control of the descent trajectory, a matter which is especially necessary for re-usable spacecraft. Missions to outer space entail increased entry speed and change the atmospheric entry conditions upon return of the spacecraft to Earth. Naturally, the need will arise to increase the available maneuver possibilities of an entry capsule, most probably by using aerodynamic lift. However, an increase in aerodynamic lift entails solution of a number of difficult structural and engineering problems: heat protection of the support surfaces of the spacecraft, accurate prediction of the trajectory, and control during the complete entry, including the section when communication with the command point is difficult because of the high degree of ionization in the air in the shock wave. Therefore, the simplest entry arrangement was chosen in the first missions for orbital spacecraft, i.e., a ballistic trajectory, and the entry vehicles took the form of capsules with zero aerodynamic lift.

Spacecraft with zero lift and controlled drag are also, in essence, ballistic entry vehicles, since control of drag allows them to change the distribution of deceleration in the trajectory, but does not allow them to maneuver and choose the landing point

in the main phase of the entry. The Soviet Vostok spacecraft is a ballistic sphere-shaped vehicle, and the American Mercury spacecraft has a capsule shape.

Spacecraft designed to create a small lift force during entry because of displacement of the center of mass (this is achieved by having the spacecraft at an angle to the flow), can perform a controlled descent. Such vehicles are semi- /249
ballistic. Examples are spacecraft of the Soyuz and Apollo type, in which the center of gravity is displaced relative to the axis of symmetry, and a positive lift force is obtained at a negative angle of attack. The main lift force component is the axial pressure force on the front surface of the vehicle. Control of such a vehicle can be accomplished both in roll, and in pitch.

Control of this kind of vehicle in roll makes it possible to perform some maneuvers in the trajectory, to compensate for error in the initial entry conditions, and to reduce the level of deceleration and heating during the entry. This refers to vehicles of nonaxisymmetric shape, for which the maximum lift-to-drag ratio is about 0.5.

Finally, promising entry vehicles with a lift-to-drag ratio of more than 1.0 — 1.5 take the form of winged spacecraft or vehicles with a lifting body. These will have wide maneuver capability. The level of deceleration and intensity of heating in the descent trajectory for such vehicles, even with initial velocities exceeding the first cosmic speed, will be comparatively low, although the total amount of heat reaching the vehicle surface during the entry will increase. During entry of vehicles of this kind, one can choose the touchdown point within wide limits, even for off-design entry conditions, and

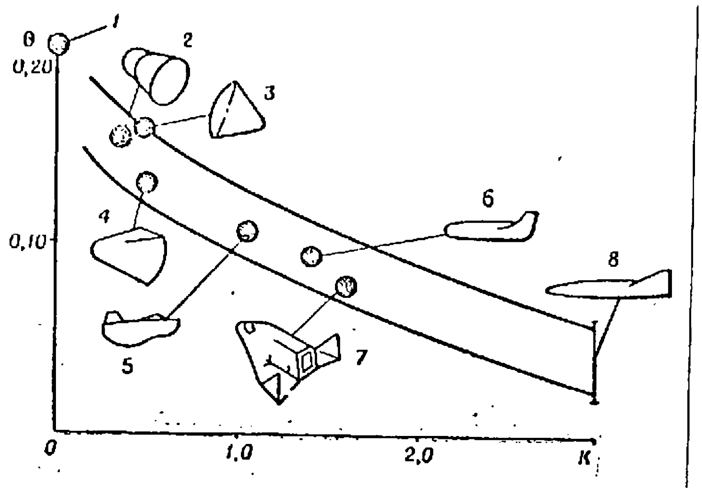


Figure 91. Comparative characteristics of the volumetric efficiency of space aircraft of different configuration, as a function of the hypersonic lift/drag ratio.

1- sphere; 2- semi-ballistic Gemini spacecraft; 3- semi-ballistic Apollo spacecraft; 4- the M1 vehicle; 5- the M2 vehicle; 6- the X-24A vehicle; 7- a development of the Dyna-Soar X20 rocket glider; 8- hypothetical vehicle with large lift/drag ratio.

accomplish an aircraft type of landing. This is the reason why the construction of vehicles in recent years with quite high lift-to-drag ratio has attracted considerable attention. Figure 91 shows the comparable characteristics of the volumetric efficiency Θ as a function of the hypersonic lift-to-drag ratio K for semi-ballistic vehicles of the Gemini and Apollo type, the experimental M1, M2, and X-24 vehicles with a blunt lifting body, and also for the X-20 rocket aircraft and a proposed vehicle with large lift-to-drag ratio. In the future, aircraft will be built to satisfy the flight requirements at hypersonic, transonic, and subsonic flight speeds.

We now discuss some of the vehicles for which designs have been considered by various foreign institutes and companies:

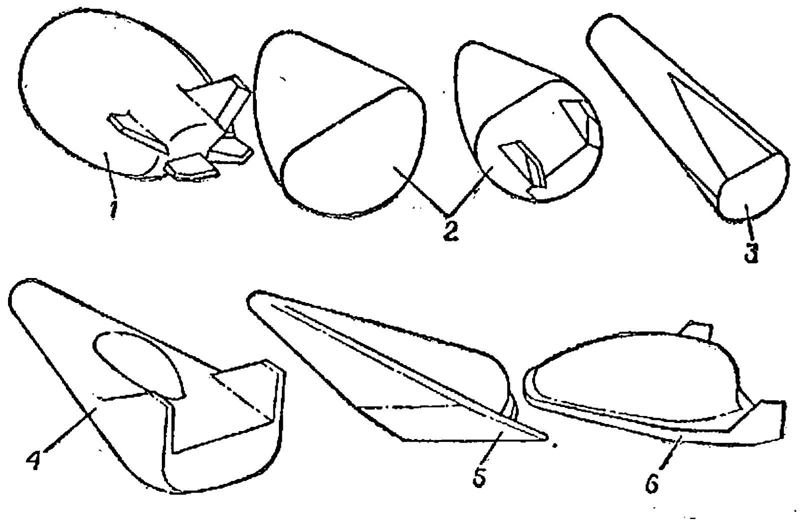


Figure 92. Shapes of lifting body vehicles.

1- oblate spheroid; 2- blunted cone with semivertex angle of 60°; 3- blunted slender elliptical cone; 4- blunted 26° cone; 5- short wing vehicle; 6- flat nose vehicle with contoured edge.

a. vehicles with a blunt lift generating body (Figure 92), having a high volume efficiency; the hypersonic lift-to-drag ratio is 1.1 — 1.3, and the subsonic ratio is 3.1 — 3.5, which makes possible a horizontal landing without using auxiliary devices to create lift after the vehicle enters the atmosphere;

b. vehicles with an elongated lift generating body (Figure 93), which have hypersonic lift-to-drag ratio of more than 3.0, allowing the vehicle to land in a given region after a descent from orbit at any time; the subsonic lift-to-drag ratio of this vehicle does not meet the requirements for safe landing, /251
and, therefore, the vehicle is designed with a moving wing to produce a lift-to-drag ratio up to 5.0 — 7.0 at subsonic flight speeds;

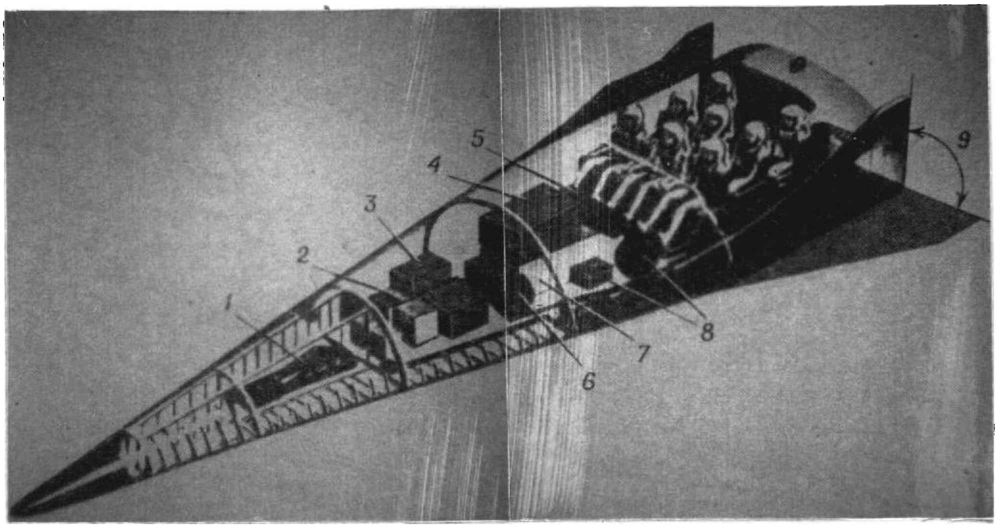


Figure 93. Vehicle with elongated lifting body.

1- nose landing shoe; 2- electric supply source; 3- navigation and guidance system; 4- storage system; 5- communication system; 6- life support system; 7- cargo compartment; 8- vehicle attitude control system; 9- elements of the variable geometry structure.

c. vehicles which use parachutes, motors, helicopter blades, or other auxiliary devices to create lift (Figure 94) during transition from the flight regime following atmospheric entry to the approach and landing regime; the hypersonic lift-to-drag ratio of these vehicles is 2.3 — 3.0, and the subsonic ratio is 1.0, so that the use of additional devices to create lift is required in this flight regime.

The USA has tested several experimental single seat aircraft in its program for promising spacecraft with lift, capable of launch and landing at ordinary airports and re-usable for shuttle flights in the Earth-orbiting laboratory-Earth sense. These vehicles (Figure 95), with a launch mass of up to 5000 kg, have a lift-to-drag ratio at subsonic speed of 3.1 — 3.5 and 1.2 — 1.4 at hypersonic speed. Usually, the vehicles have liquid rocket motors, operating with liquid oxygen and a mixture

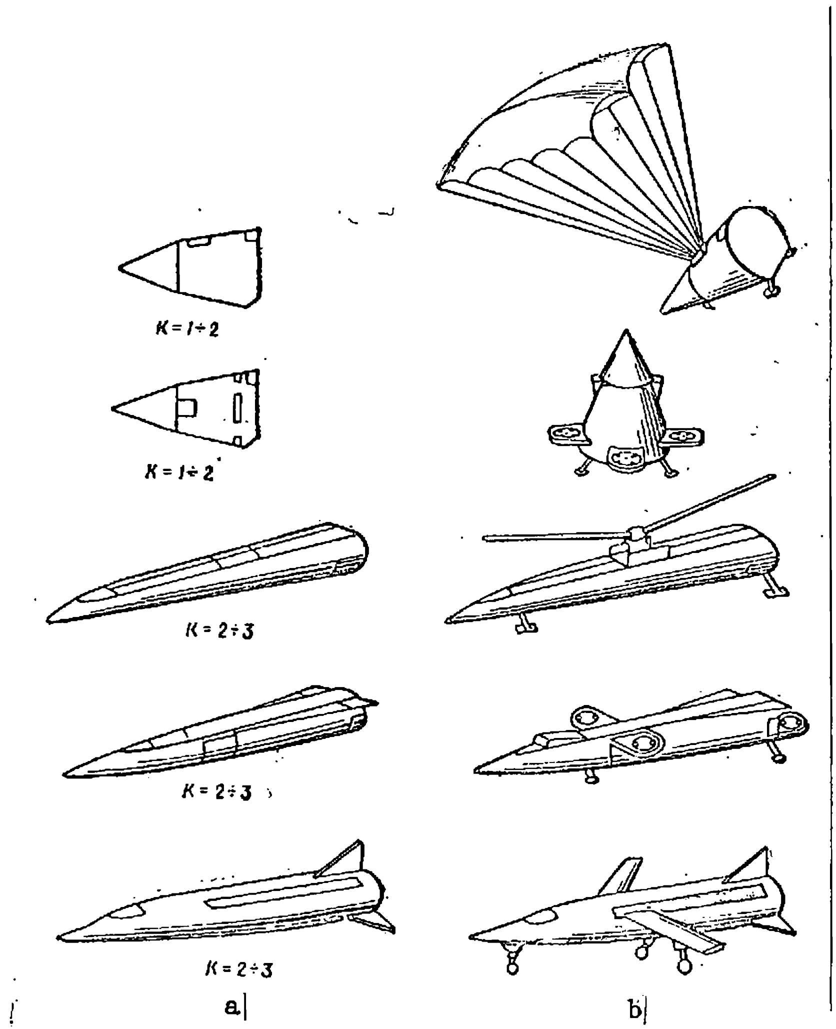


Figure 94. Diagrams of vehicles with auxiliary flight devices. a- space vehicle configuration for atmospheric entry; b- vehicle configuration in the landing regime.

of ethyl alcohol and water, or an oxygen-hydrogen motor. Some of the vehicles also have an auxiliary motor, consisting of several liquid motors operating with hydrogen peroxide. The additional motors are designed for control of the vehicle in the final section of the descent and landing and are used at the discretion of the pilot.

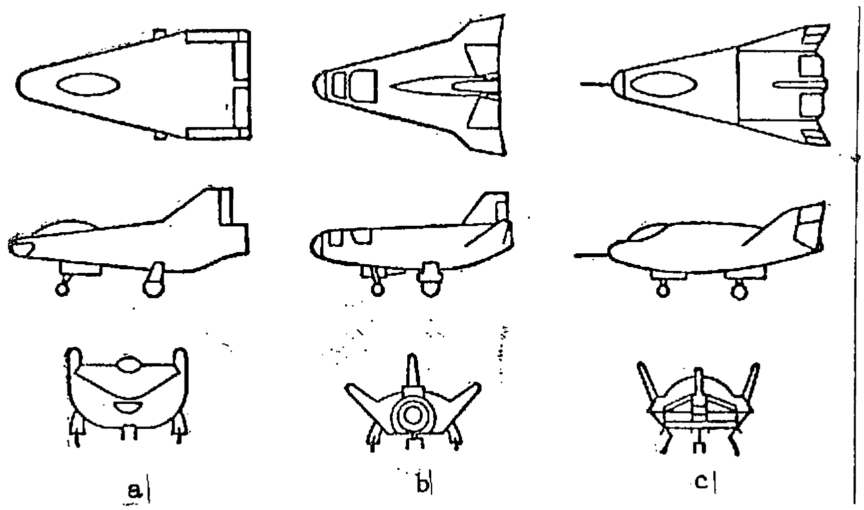


Figure 95. Comparative diagrams of experimental vehicles.
a- M2-F2; b- HL-10; c- X-24A.

The vehicle control system in atmospheric flight consists of aerodynamic control surfaces, mounted on the stern of the vehicle.

FIRST EXPERIMENTAL VEHICLES

One of the first manned lifting body vehicles was the M2-F1 vehicle, which was towed by an aircraft, and then separated at a height of about 4000 m, and made a landing at an airfield after a flight of three minutes. Its speed in flight was 217 km/hour, and its landing speed was 130 km/hour. This vehicle made many flights which confirmed that it was possible to build a lifting body vehicle with enough aerodynamic lift and control to perform gliding flight, maneuvers, and a landing.

The foreign press has reported that another vehicle (M2-F2) has made gliding flights at subsonic speed, using a liquid rocket motor, and studies were made of the stability characteristics,

the effectiveness of the control surfaces, the flying characteristics, the aerodynamic loading, and the control and maneuverability in approach and landing. Because of poor lateral-longitudinal control, the vehicle had an emergency in one of its flights (the pilot lost control because of large amplitude roll oscillations at low altitude, and was not able to use the auxiliary motor and lower the undercarriage). /254

Subsequently, more sophisticated experimental lifting body vehicles were built (the HL-10 and the X-24A).

The experimental HL-10 vehicle had aerodynamic control surfaces located on the rear. In addition, to increase the vehicle stability, it used six additional surfaces which were set in the required position by means of electric motors. The auxiliary motor also consisted of a liquid fuel engine operating with hydrogen peroxide.

The maximum lift-to-drag ratio of the HL-10 at subsonic speed was 3.3. During its first gliding flight, after separating from the towing aircraft at a height of about 14,000 m, the vehicle made a landing in a time of 3 minutes, 7 seconds.

The manned X-24A vehicle (Figure 96) had a lift-to-drag ratio of 1.2 — 1.4 in hypersonic flight, and about 4.5 in subsonic flight.

The vehicle was of semimonocoque structure, made of aluminum alloy and stainless steel. Like previous vehicles, the X-24A has a flat lower surface, and the contoured upper side of the nose takes the form of a circular arc to provide longitudinal stability at transonic speeds. The vehicle is designed for a dynamic head of up to 2400 kG/m^2. Its rear part

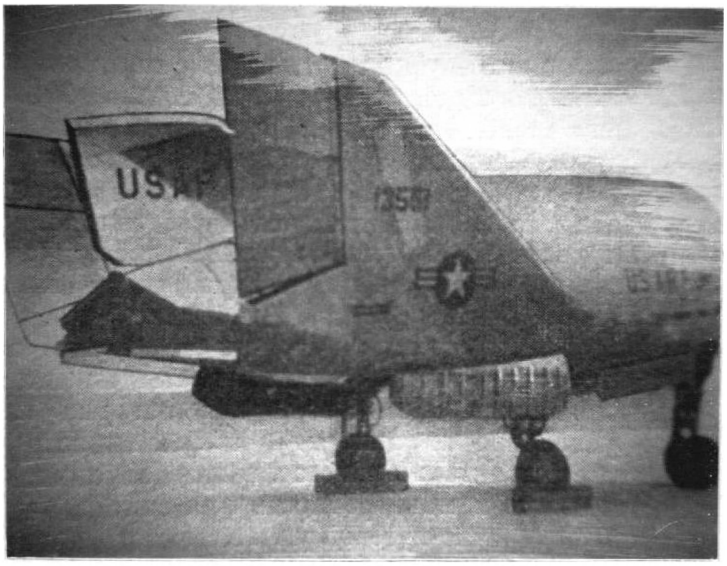


Figure 96. The X-24A experimental vehicle.

carries three vertical fins and eight moving aerodynamic control surfaces: two upper and two lower panels and split rudders on the two external vertical panels. The configuration is capable of being changed somewhat, depending on the flight regime, by inclining the control surfaces.

Beginning in 1969, this vehicle made gliding flights, after separating from an aircraft at a height on the order of 14,000 m. Later on, it carried a liquid rocket motor with which it could reach a height of 30 km, and a speed corresponding to Mach number M = 2.0.

According to press reports, the X-24A reached a height of 20 km and a speed of 1430 km/hour in one of its flights. The flight lasted seven minutes, reckoned from the time of release from a B-52 aircraft. This flight apparently completed the X-24A lifting body vehicle program. During the program, 28 flights were performed. According to comment by the pilots, the special features of flying the X-24A experimental vehicle was the difficulty in controlling it in the flight section and

in the landing, since the average speed of vertical motion was 70 m/sec (compared with 25 — 35 m/sec for contemporary aircraft on a standard approach and landing).

Later, a modified lifting body vehicle was built (the X-24B), which, according to foreign press reports, made its first flight on August 1, 1973. The vehicle was released from a B-52 aircraft at a height of 12 km, and descended in a gliding flight and landed. The flight lasted for four minutes. This vehicle was the last in the lifting body series, used to demonstrate the possibility of maneuvering and safe landing. It had a shape basically suitable for space flight, and also for future hypersonic aircraft, whose cruising speed would be about 5600 km/hour.

There is a report that, in 1973, a NASA research center proposes to build a new modification of the lifting body vehicle (X-24C), which would differ from the others of this type by having a high power liquid rocket motor (thrust 6800 kG).

The final objective of this program of lifting body vehicle tests, as mentioned above, is to simulate the terminal phase of flight and landing of an orbital stage, on its return from a space flight. The flight characteristics of the M2, HL-10, and X-24A are limited by the lower limit of velocity head (244 kG/m^2), which limits the efficiency of the control surfaces, and the limit imposed by the structural strength (1950 kG/m^2). The information obtained during flight tests of the HL-10 will be used to design high speed aircraft and re-usable air-space vehicles.

In the tests, the chief interest was to investigate control systems for the orbital stage of an air-space vehicle, particularly to investigate a jet control system. The motivation is that lifting body vehicles must be equipped with this kind of system in order to carry out maneuvers in space. /256

An important task in these tests was to demonstrate the possibility of performing manned flight without using a power unit. The vehicle must possess favorable flying and engineering characteristics and a minimum subsonic lift-to-drag ratio (somewhat more than 3) in order to accomplish successful flight in the desired region, level off, and land. For the X-24A and HL-10, this value was 4.6, which was greater than the minimum subsonic lift-to-drag ratio required for an unpowered landing of a lifting body vehicle. However, the value determined in wind tunnels and in flight experiments was not confirmed. It turned out to be very close to the minimum value for which these vehicles could be landed without a power unit.

SOME RE-USABLE SPACE VEHICLE DESIGNS

The tests and investigations of lifting body vehicles enables us to study their possible use as re-usable multi-passenger and transport space vehicles.

One design for this vehicle is the one and one-half stage arrangement with a lifting body of variable sweepback, to the sides of which fuel tanks are attached, with liquid hydrogen and liquid oxygen to supply the motor, which is located at the rear. The mass of the fully loaded vehicle is 272 tons. The vehicle is injected into orbit by motors developing a thrust of 340 tons, and can carry a payload of 11.35 tons to a height of 185 km. To improve the lift-to-drag ratio of the lifting

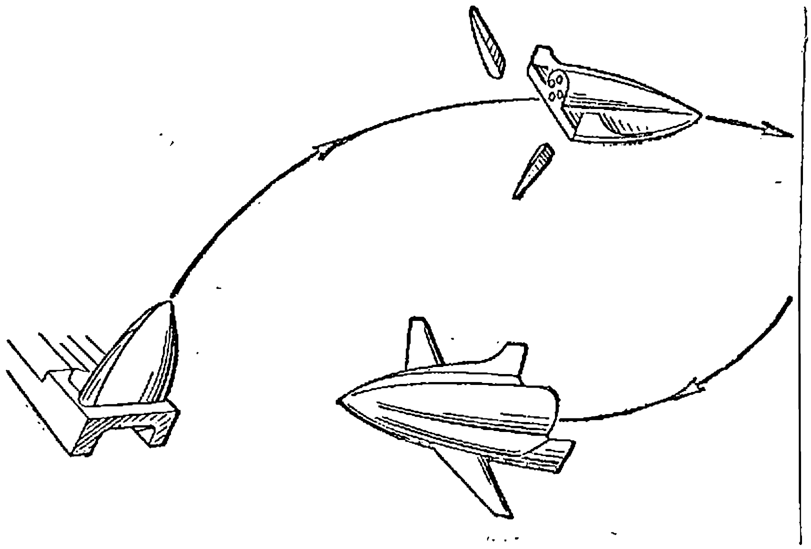


Figure 97. Illustration of re-usable space vehicle.

body vehicle, there are wings which are folded back at launch and are deployed during the approach and landing. Having wings allows the vehicle to use the principal airports. The vehicle must be launched vertically. After the propellant is used, the external tanks will be jettisoned and it goes into the design orbit, and following completion of the planned operations, it returns to Earth using the movable wing (Figure 97) during subsonic flight. The plan is to use a launch vehicle to launch this vehicle.

A lifting body vehicle is planned which is mounted in the middle of a V-shaped fuel tank. It has a body with a flat bottom and a sealed compartment for the crew and passengers, a cargo compartment, an engine and fuel tanks, and systems and equipment for flight to space and Earth return. For subsonic flight, it uses a movable wing. The hypersonic lift-to-drag ratio is about 2.0. The engine consists of three liquid fuel motors with high pressure chambers using liquid hydrogen and liquid oxygen, and developing a vacuum thrust of 160 tons each.

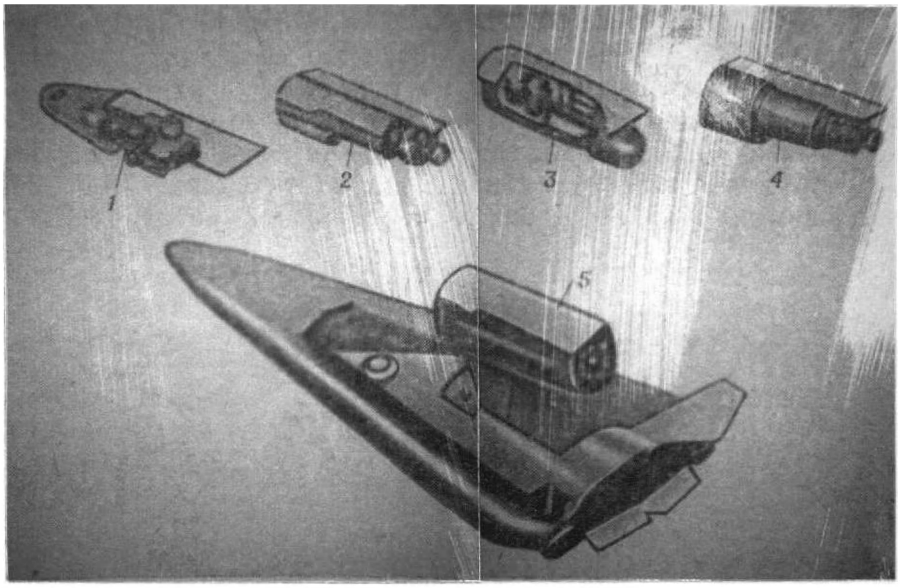


Figure 98. Layout of multipurpose space vehicle.

1- load mass 3290 kg, personnel weight 570 kg; 2- load mass 9026.5 kg; 3- load mass 5350 kg, personnel mass 520 kg; 4- load mass 9980 kg; 5- standard cargo module.

The fuel tanks hold about 23 tons of propellant. The propellant in the jettisonable fuel tanks is used in the motors to inject the vehicle into orbit. The necessary velocity increment to reach the design orbital speed, perform maneuvers in orbit, and a retroburn to descend from orbit are provided by the motor, which uses propellant from the tanks located inside the vehicle body. The vehicle has a system of aerodynamic control surfaces, and a conventional undercarriage of three-wheel type. The vehicle is launched vertically, and lands horizontally.

The vehicle structure is of such form that it can take cargo modules of various weights and volumes. It is proposed to use contemporary technology to construct a vehicle capable of 20 to 25 launches (Figure 98).

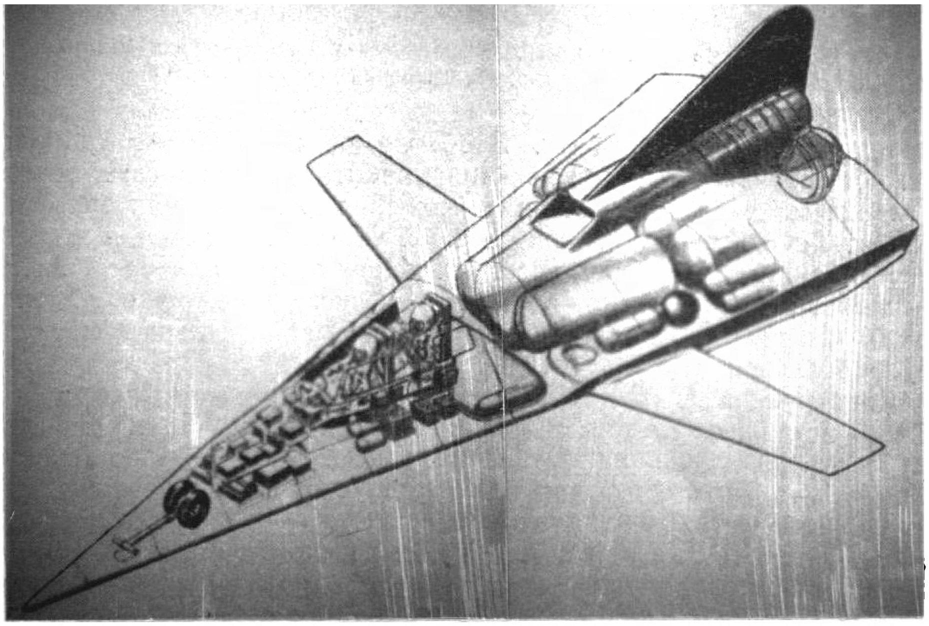


Figure 99. Layout of multipurpose re-usable space vehicle.

One such vehicle for flight to the limits of the atmosphere uses a solid fuel motor, and for flight in space, it uses a rocket motor operating with hydrogen-fluorine propellant. To have a small landing speed and a crew of two, this vehicle should have a speed corresponding to M = 20. The solid motor propellant is located in a fuel tank positioned behind the crew cabin, and the solid motor air intakes are ahead of the rudder at the rear (Figure 99). The rocket motor is located at the rear below the solid motor.

Also planned is an air-space system (Figure 100), consisting of three re-usable vehicles, two (outer vehicles) being used for launch, and one as the orbiter. Each vehicle has a liquid motor. Each vehicle has a crew of two. The system is launched vertically. The two launch vehicles carry propellant, which is used in the motors of all three vehicles during the launch.

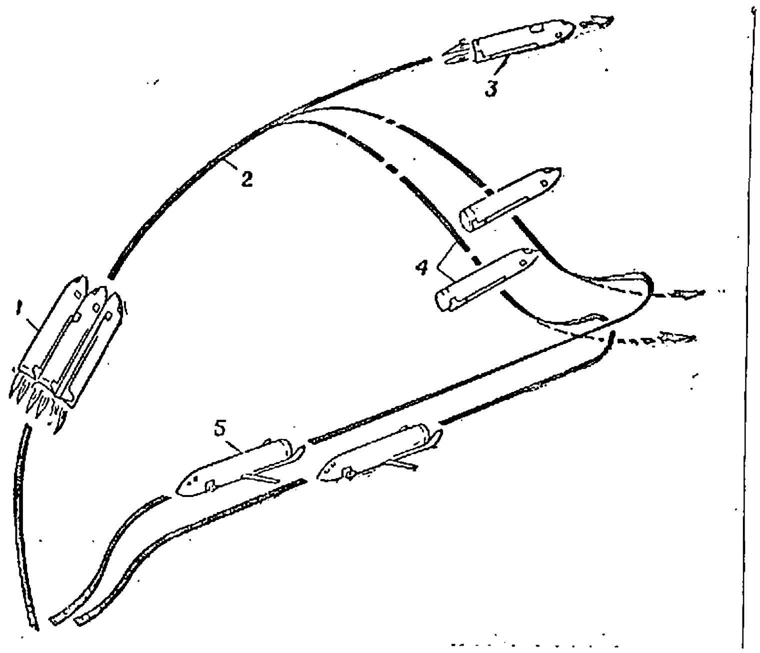


Figure 100. Illustration of flight of space-air system.

1- vertical launch of the vehicle; 2- separation of launch vehicles from orbiter; 3- orbiter in orbit; 4- re-entry of launch vehicles; 5- return of launch vehicles to the launch site area.

On reaching a speed of about 2500 m/sec, the launch vehicles separate from the orbiter, and using their lift, a movable wing with variable sweepback, and a turbofan engine, they descend and perform a controlled horizontal landing near the launch pad. The orbiter uses its motor to fly on into the designed orbit, and then, having finished its flight program, descends from orbit, enters the atmosphere, and lands horizontally, like the launch vehicles.

One design, in which the launch is performed with the help of a launch vehicle, consists of the following elements: /260

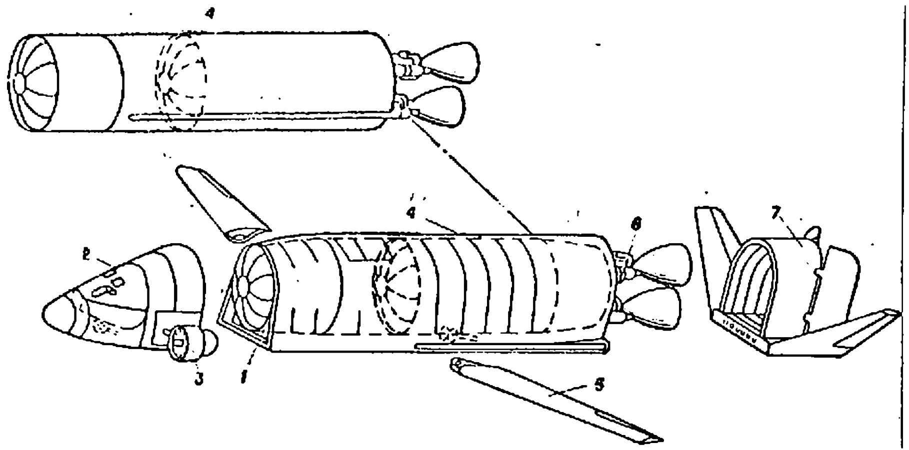


Figure 101. Basic elements of vehicle with mixed propulsion units.

1- cylindrical section; 2- nose section; 3- movable turbofan engine; 4- cylindrical section of launch vehicle; 5- movable variable geometry wing; 6- liquid fuel motor; 7- base section with aerodynamic control surfaces.

a) a cylindrical section to which are attached a nose and a tail (Figure 101); the nose region carries the command point with the astronaut cabin and a fixture for docking with the orbiter; the cylindrical section houses a motor, fuel tanks, and a compartment to accommodate twelve people; the tail section carries the aerodynamic control surfaces;

b) two liquid motors of thrust 113 tons each, to which propellant is supplied by a high pressure force system;

c) two turbofan engines started in flight, for flight at subsonic speed during the Earth re-entry;

d) a movable wing to provide the required aerodynamic characteristics at subsonic speed; this system should inject a payload of more than 11 tons into heliocentric orbit with a small angle of inclination to the equatorial plane, or a payload of more than 8 tons into polar orbit.

A TWO-STAGE SHUTTLE VEHICLE DESIGN

A new design has been reported comparatively recently in the press abroad for an air-space re-usable vehicle, intended for transport of passengers and supplies in the Earth-geocentric orbit-Earth route. The proposed arrangement (Figure 102), as reported, has been investigated abroad in various organizations, and is reminiscent of a subsonic aircraft consisting of two re-usable stages. The vehicle launch mass is tentatively 1135 tons, and the mass injected into orbit is 11.3 tons. Each stage is equipped with an engine generating a speed up to 500 m/sec in vacuum, and is an efficient vehicle when entering the atmosphere. The orbiter stage is a space vehicle, has a motor for maneuvering in orbit, a life support system, a thermal control system, and all that is necessary to stay in Earth orbit from 7 to 30 days. /261

During launch, the second orbit stage is located in the carrier (first stage). Both stages use liquid oxygen and liquid hydrogen as propellant. Immediately after separation, the carrier enters the atmosphere, its speed decreases, and it transfers to subsonic flight. Then the carrier cruises at subsonic speed, and as can be seen from the schematic (Figure 103), performs a flight with a range of about 550 km in the launch region, and lands. The layout of the orbiter stage is shown in Figure 104. The large size of the orbital stage stems from the provision of large volumes for propellant and a large cargo compartment.

The cryogenic propellant, expended during maneuvers of the orbiter after orbit injection, is located in two separate tanks, which improves the storage conditions. In contrast with an ordinary aircraft, this vehicle performs an efficient

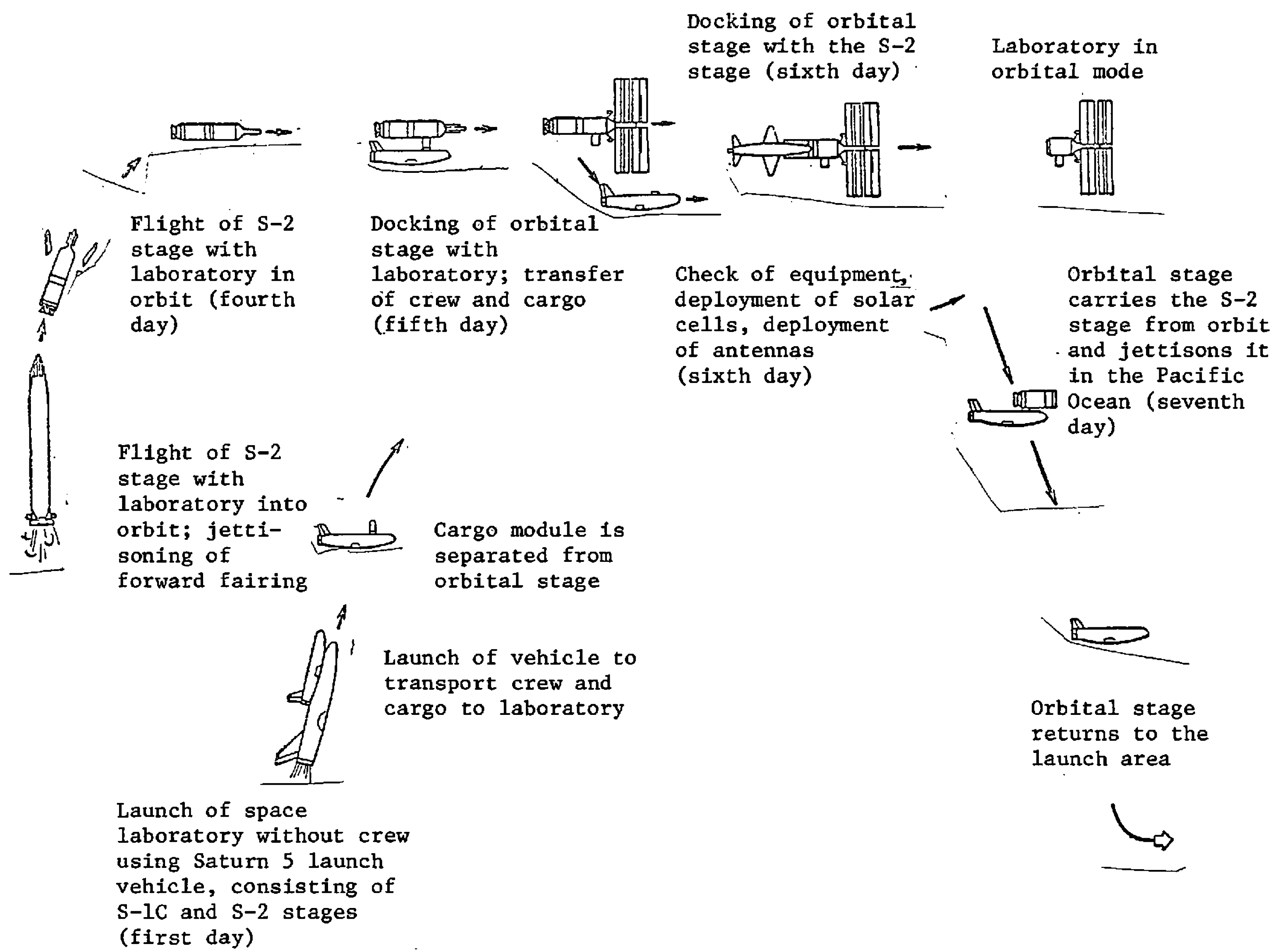


Figure 102. Illustration of deployment of space laboratory using two-stage air-space vehicle.

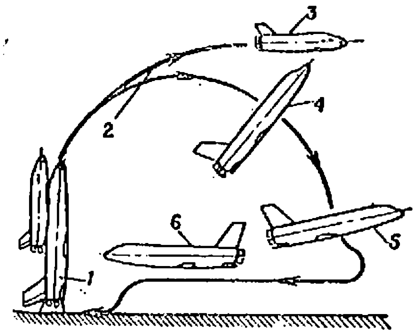


Figure 103. Illustration of use of orbiter carrier.

1- vertical launch; 2- stage separation; 3- orbiter; 4- atmospheric entry of carrier; 5- start of transfer to horizontal cruising flight at height 15,000 m; 6- return of carrier to the launch area using solid fuel motor.

atmospheric entry from the viewpoint of heating, stability, and controllability. During atmospheric entry, the orbiter has a very high angle of attack (about 60°), at which it creates large lift and drag, the lift-to-drag ratio being about 0.5. The orbiter must maintain its attitude during the entire descent to 12,000 m. At this height, the speed of descent is reduced to 90 m/sec and even less, and then the angle of longitudinal inclination is also reduced and the stage adjusts its attitude until the speed drops below the value for which horizontal flight is possible. Then the solid motor is ignited and the stage performs a descent, approach, and landing, like an ordinary aircraft. The foreign press reports that investigations have shown that there are advantages in atmospheric entry with maximum or close to maximum lift coefficient. Atmospheric entry of the orbiter near the maximum lift attitude begins almost at maximum height and, therefore, the heat flux is reduced, as well as the equilibrium radiation temperature. The large drag then arising reduces the length of the heat pulse. Therefore, the temperature of the skin and the amount of heat shield below the skin can be minimized.

At large angle of attack, only the lower surfaces of the stage are exposed to the effect of the oncoming flow, and the area of these surfaces represents roughly one-third of the wetted surface. The remaining surface of the stage is located /263

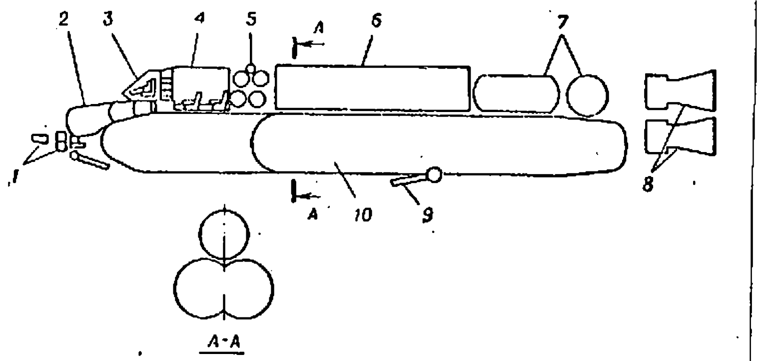


Figure 104. Layout of orbital stage.

1- guidance system;, electronic equipment, and electrical supply source; 2- solid fule landing motors; 3- crew cabin; 4- passengers; 5- gas tanks for jet attitude control system; 6- cargo; 7- propellant for use in orbit; 8- main motors; 9- undercarriage; 10- fuel tanks for main motors.

in a low density flow, or, what is more important, in a region of flow separation. Therefore, the greater part of the stage surface (the top surface of the wing, the horizontal tail assembly, and the body) do not require a heat shield layer, while the other surfaces (e.g., the side walls of the body) require maximum heat protection from aerodynamic heating. The regions above the wing and the body are ideal areas for locating equipment sensitive to high temperature (e.g., radiators glass cabins, and solid motor gondolas).

The simplest device for control of the orbiter stage is a jet system. It must be mounted on the stage for control of attitude in space and for carrying out transfer maneuvers in space. If additional propellant is carried, it can also be used in atmospheric entry.

The lower surfaces of elements of the orbiter stage are areas which create considerable aerodynamic lift, and, therefore, the pressure on these areas mainly determines the stability and balance during entry of the orbiter into the atmosphere. The transverse stability is achieved by the dihedral angle of the lower surface of the wing. It is calculated (and clearly, not without foundation) that it is important to make the bottom of the nose part of the body stage sufficiently wide and flat, to avoid the occurrence of destabilizing moments, which could arise from side forces acting on the nose region. Stabilizing forces can be created by giving the stabilizer dihedral.

The sequence of operations in the vehicle descent should be noted in the above. First, the orbiter descends with an angle of attack of about 60° to a height of 12,000 m, then it transfers to horizontal flight at subsonic speed. Since the body area of the orbiter at the initial stage of descent is large in comparison with the wing area, it generates the main part of the lift. After the transfer maneuver, almost all the lift is created by the wing. Therefore, in the transfer, the /264
orbiter acquires a considerable speed and loses about 4500 m of height. The gas turbine motor must be fired to acquire a steady glide condition and the stage then carries out a normal approach and landing. The wing has to provide cruising flight at subsonic speed and the landing maneuvers. Since the atmospheric entry is made at large angle of attack, the wing planform must be chosen to optimize flight in the cruise and landing regime. In this case, a straight wing with sufficient taper is better than any other wing; it provides the required lift at suitable landing speeds, and is also an ideal wing for subsonic cruising flight. Atmospheric entry at large angle of attack eliminates the need for a movable wing.

The angle of attack of 60° during descent is a compromise. For a vehicle in equilibrium at this angle, the duration of aerodynamic heating and its intensity are quite small, and the maximum gravity loading does not exceed 2 g. This is quite suitable for passengers who do not require special training before a space flight. The lift-to-drag ratio of the orbiter at hypersonic speed at this angle of attack is a little over 0.5. The maximum lateral range of this vehicle during atmospheric entry is 400 km. It is suggested that the lateral range can be increased by increasing the heat shield mass. In this case, the leading edge of the wing, the body, and the fin will experience more intense heating as the lift-to-drag ratio and the lateral range are increased. The duration of the heat pulse increases, but until the maximum lift-to-drag ratio is reached, the total amount of thermal insulation required will be roughly proportional to the square root of the heat pulse length. Hence, it follows that an orbiter with large lateral /265 range requires more heat resistant materials and an increased heat shield mass.

The foreign press reports that models of the arrangement described have been tested in wind tunnels, to determine the optimal angles of attack for the trim position and other parameters associated with the heat flux. We note that the heating is distributed nonuniformly over the wing span because of interference with the flow over the body, which can produce local increase of heating. However, the tests and calculations could not determine the heat transfer to small local sections, whose temperature could possibly exceed 1200° C. In planning a series of launches, they propose first to perform separate launch (horizontal and vertical) of the carrier and the orbiter, and later a vertical launch of the combined vehicle (Figure 105).

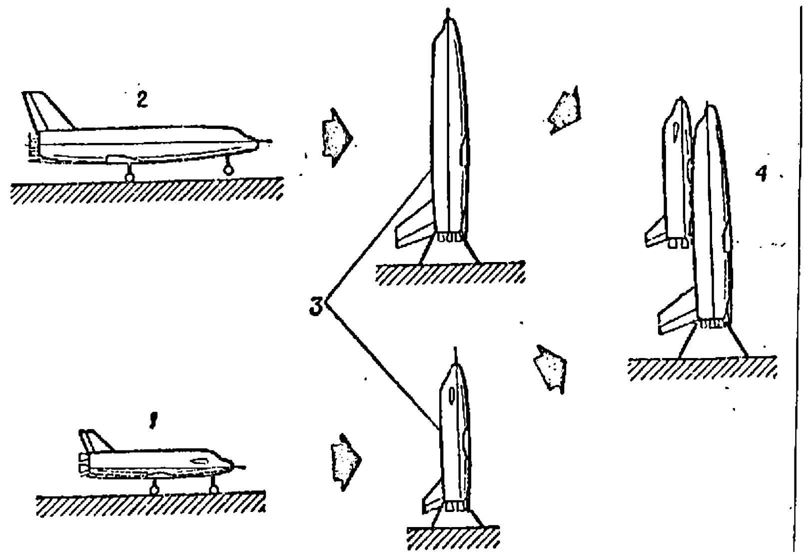


Figure 105. Sequence of flight tests for carrier and orbiter stages.

1- horizontal flight of orbiter; 2- horizontal flight of carrier; 3- vertical flight of orbiter and carrier; 4- vertical flight of space-air vehicle.

Each stage of the vehicle is designed to land horizontally with a speed of about 235 km/hour with flaps down, and with 15% margin relative to the stalling speed.

The subsonic lift-to-drag ratio of the orbiter is 8. The solid motor must produce a thrust factor of 0.3. The assumption is that these parameters will provide the basic flight and engineering requirements necessary for launch and landing at existing airfields with long runways. For ordinary flights to the orbital stage, there must be a minimum margin of propellant, determined only by having to go around again during landing. The maximum fuel stock will allow flight to a distance of up to 1000 km. /266

The space transport system. One of the USA scientific laboratories has proposed a space transport orbiter system using an orbiter with a delta-shaped lifting body. The proposal is that it will satisfy the requirements as to maximum lateral range while providing limited payload. Only the method of atmospheric entry is varied when this kind of orbiter is used.

The main requirement in the development of this system is to reduce the cost of placing cargo in Earth orbit. The cost of injecting a payload into low geocentric orbit, according to press reports, is 1100 — 1200 dollars/kg, and into synchronous equatorial orbit, it is 11,000 — 22,000 dollars/kg. The scientists and engineers who have developed these systems proposed to use hypersonic transport aircraft with air breathing motors. Figure 106 shows values of the volume efficiency coefficient ($V^{2/3}/S_W$) of space and air vehicles, existing and under development (where V is the vehicle volume, and S_W is the wetted vehicle surface area), as a function of the effective range of these aircraft, defined as the product of the Mach number M and the maximum lift-to-drag ratio K. As can be seen from the graph, future values of the volume efficiency for both space and atmospheric vehicles lie in the region of large values of effective range.

There is also a proposal to use an orbiter with a delta lifting body to make up a one and one-half stage system with jettisonable tanks. A scale model of the orbiter is shown in Figure 107.

In order to demonstrate the advantages of an orbiter with a delta lifting wing (Figure 108), it has been compared with an orbiter of ordinary type, having straight wings. It is assumed in choosing the aerodynamic layout of the orbiter that

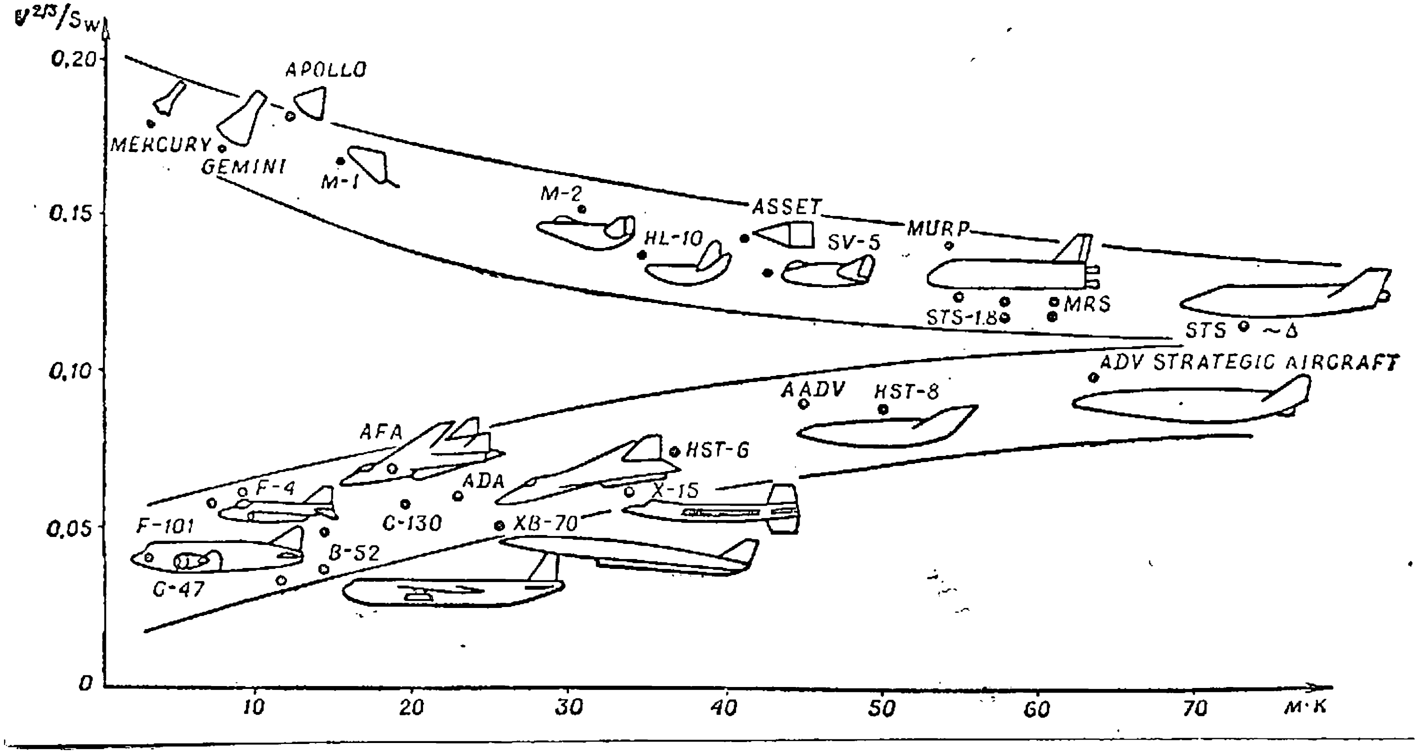


Figure 106. Graphical representation of variation of volume efficiency coefficient $V^{2/3}/S_W$ and effective range MK for some vehicles.

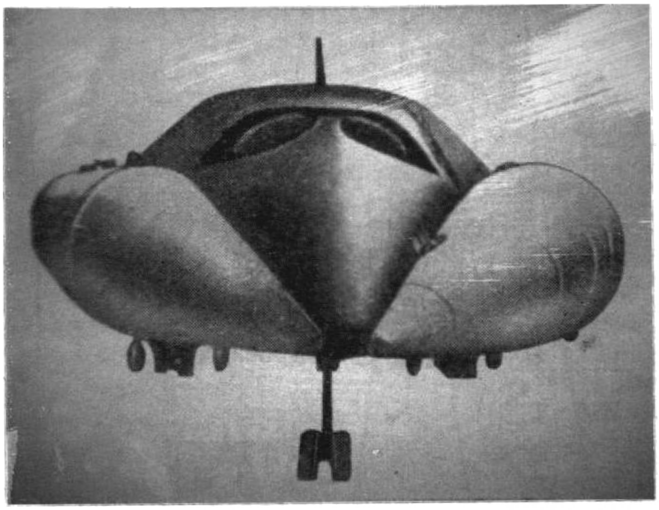


Figure 107. Scale model of orbiter.

the delta lifting body is required only for providing large lateral range. However, the range of values of maximum lift-to-drag ratio of orbiters of different types with a delta lifting body is 0.8 to 3.0, while for large angles of attack, it is 0.45 to 0.78. Therefore, depending on the aerodynamic layout of the lifting body vehicle and the type of trajectory, the lateral range can vary from 370 to 7400 km. In comparing the delta and the straight wing, we take into account their geometric size. For small size, the straight wing is preferable. The landing characteristics of an orbiter with straight and delta wings are shown in Figure 109.

In considering the design of an orbiter, one has to study problems of atmospheric entry at large angles of attack, and it is known that an increase in an angle of attack during atmospheric entry leads to reduction in the total entry time and a decrease in the integrated heat flux. Therefore, a heavier heat shield is required for large lift-to-drag ratio and small angles of attack. Consequently, when the lateral range is increased, the payload is reduced considerably because of increase in the heat shield mass. Figure 110 shows lateral and longitudinal range figures obtained for an entry with α = 50° and different speeds of lateral maneuver at small angles of attack.

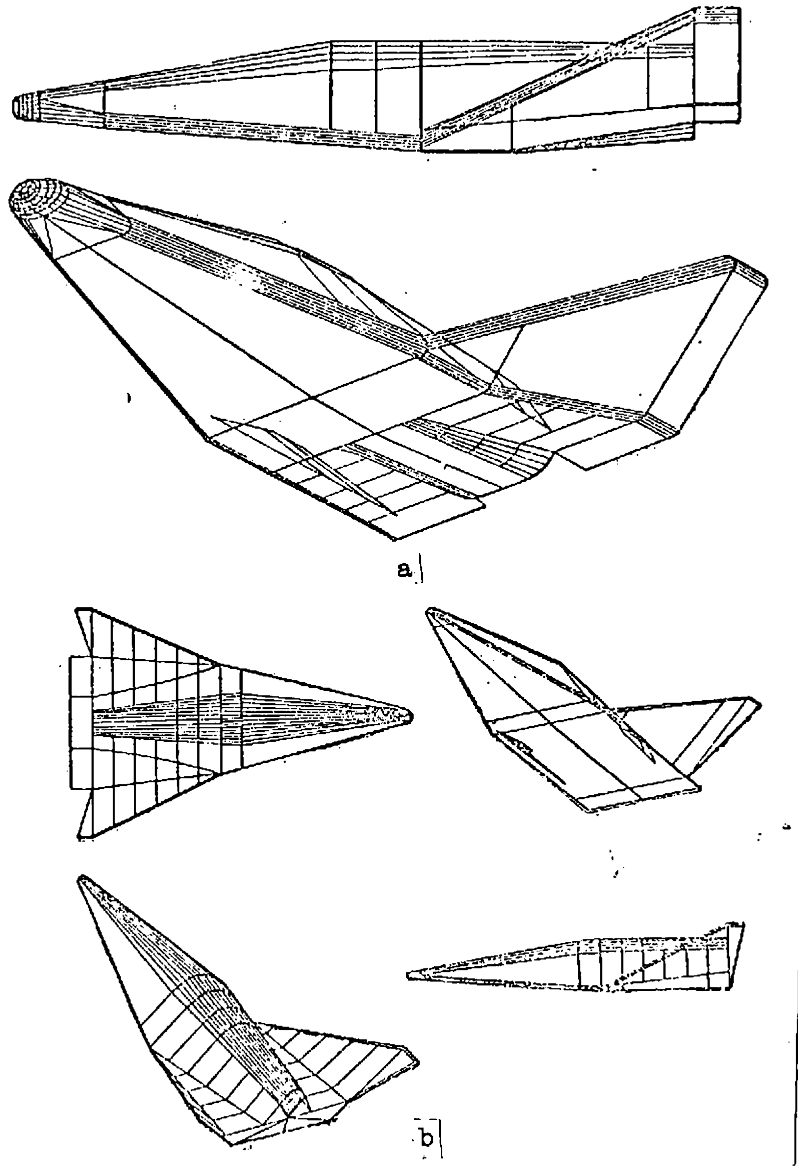


Figure 108. Diagram of delta wing orbiter.
a- K = 2; b- K = 3.

Figure 108 shows two orbiter designs with a lifting delta wing. The first design, with a mean lift-to-drag ratio of K ≈ 2, gives a limiting lateral range of 2780 km. The second with K ≈ 3 is preferable, since in an ordinary entry, it gives a lateral range of 6500 km, which can be reduced to 2780 km by

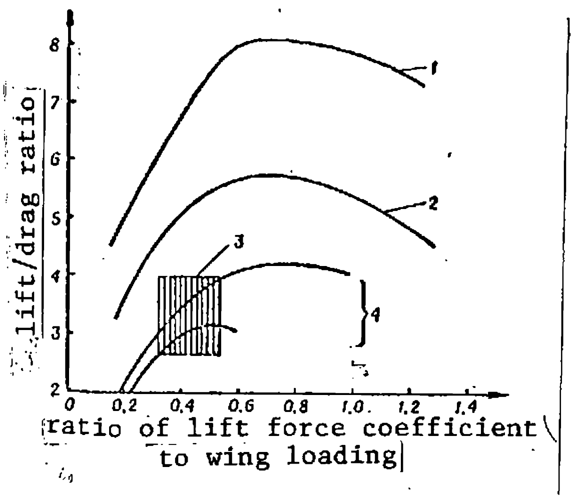


Figure 109. Landing characteristics of orbiter.

1- straight wing orbiter; 2- delta wing orbiter; 3- favorable landing characteristics; 4- existing lifting body vehicles.

using a lateral maneuver with a variable angle of attack. Here the specific heat flux is less than for an orbiter of ordinary /269 type with a straight wing.

It is assumed that the second arrangement is also stable and controllable over the whole range of angle of attack and in all the flight regimes examined, including the transonic flight and landing. The fuselage volume in this arrangement is sufficient to hold the payload and the propellant. Pitch control during hypersonic flight is accomplished by /271 means of end panels. At M = 3, pitch control is produced using the upper panel, the lower panels remaining fixed. During transfer to transonic and subsonic speeds, the upper fins remain effective. Directional control of roll, and to some extent in pitch, is achieved by means of end panels on the wings.

With increase of angle of attack, the temperature of the shell of a delta wing increases more slowly than for a straight wing. Therefore, the arrangement with a delta lifting body has more uniform temperature distribution, and an interference zone does not occur here. The straight wing arrangement has high temperature regions with α = 40° and α = 60°, and interference zones for α = 40°.

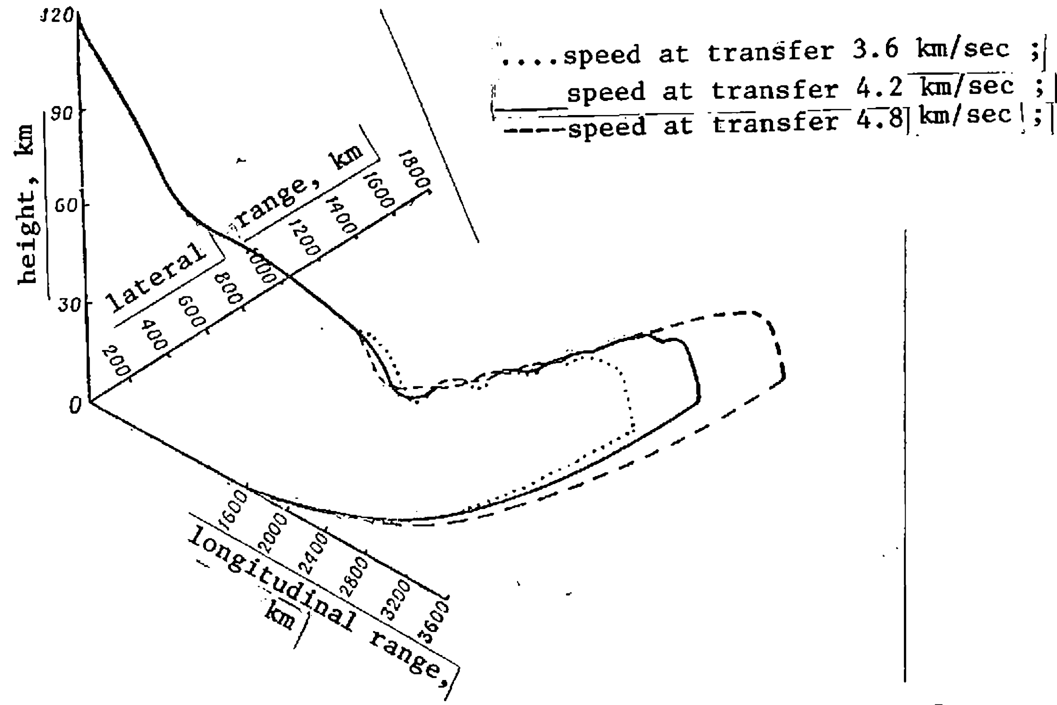


Figure 110. Orbiter flight range for entry angle α = 50°.

It is proposed that the "hot" load carrying structure will include high temperature heat resistant alloys, but the thermally insulating tanks of cryogenic fuel cannot support loads. It is also proposed that the delta wing orbiter will have a titanium "hot" structure with a temperature limit of 315° C. It has been mentioned that ground and flight tests, as well as numerous investigations conducted during the last 15 years, have demonstrated the high reliability of the delta wing orbiter, but the question of heat protection remains unresolved. The success of the entire program depends to a considerable extent on these problems being solved. In particular, the possibility of using nonmetallic materials should be evaluated; these hold out promise both with regard to the weight reduction, and also as regards cost. Since the heat shield has a large area, even a small reduction of mass of heat shield (by 0.5 to 1.0 kg/m^2) will give considerable saving of mass for the orbiter. Nonmetals

are being investigated for heat shield systems using ablation, and for passive heat shields. The main objective of the program to develop ablation materials is to reduce the cost, since their reliability in use, as has been mentioned, was verified in the Apollo program. Cost reduction is particularly important, since ablation materials are intended for single use.

One defect of nonmetals is the high porosity, which stems from the need to reduce the mass volume. Because of porosity, they absorb moisture strongly and are subject to erosion in flight. Special coatings are used to improve the properties and increase the emissivity of nonmetals. In general, a heat shield system fabricated with nonmetals is an insulating layer, superimposed on the outer surface of the orbiter. The advantage of this heat shield system is that it does not require additional internal insulation, such as is necessary when metal screens are used.

Work is continuing to develop metallic materials for heat shield systems, based on solid alloys, capable of withstanding temperatures of 980 — 1100° C, and a special alloy with a niobium coating for even high temperatures. The expected temperatures for the launch and orbital stages of reusable spacecraft are shown in Figure 111.

There is a program to investigate nonmetallic materials of ceramic type, based on silica, mullite (aluminum, silicon oxide) and a combination of zirconium fibers with these materials. The addition of zirconium increases the useful temperature of the material, but it increases the mass and the cost. It is suggested that materials based on silica and mullite are capable of long term operation at a temperature of 1100° C and short term operation at 1650° C, while zirconium based materials are

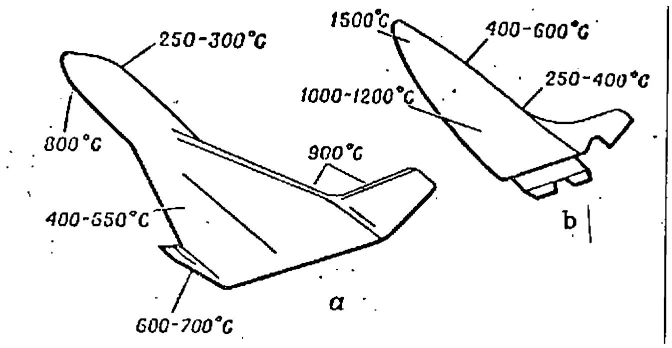


Figure 111. Expected temperatures for vehicle stages.
a- launch stage; b- orbiter stage.

capable of long term use at temperature greater than 1650° C. These materials are proposed for use on the entire surface of the orbiter, although their operating temperature properties are higher than is necessary for the major part of the surface.

Engineers abroad consider that the use of nonmetallic materials in the fabrication of orbiters with low lateral flight range will enable the system heat shield mass to be reduced by 3627 — 4535 kg, compared with a system using a metallic radiative screen.

When nonmetallic materials are used, there is no problem associated with thermal expansion, particularly for temperatures above 1100° C (the high expansion coefficient of metal requires development of special devices that allow expansion of the structure when heated and avoids buckling of thin shells). Thus, broad investigations are being conducted both into metallic and nonmetallic materials and systems to evaluate their heat shield properties in orbiters.

The choice of metal for a heat shield system is made on the basis of cost, which depends to some degree on the orbiter geometry and the lateral range (the heat shield mass is known to increase with increase of lateral range). Compared with nickel based alloys, nonmetals have greater potential capability for cost reduction. The choice of range and the associated vehicle geometry (small lateral range means a straight wing, and large lateral range means a delta wing) is evidently conclusive, although it has not yet been made, but some designers consider that preference will be given to an orbiter with a large lateral range, capable of landing over a wider range of conditions.

The press abroad carried a report that, at the start of the seventies, NASA concluded its contracts with American companies to conduct work aimed at creating a two-stage transport vehicle for journies between the Earth and geocentric orbits. Several firms presented designs for this spacecraft during the preliminary competition.

Figure 112 shows two designs. Both of these used the same configuration for the first stage, and had different second stages. In one case (Figure 112a), the second stage had a delta wing, giving wider maneuverability in the lateral direction in the atmospheric descent, and had tapered stabilizers, mounted at an angle of 15° to the vertical; the angle of attack of this stage during atmospheric entry was 55°, and 20° after atmospheric entry. In the second case (Figure 112b), the second stage had a straight wing, and is capable of only limited maneuvers in the lateral direction.

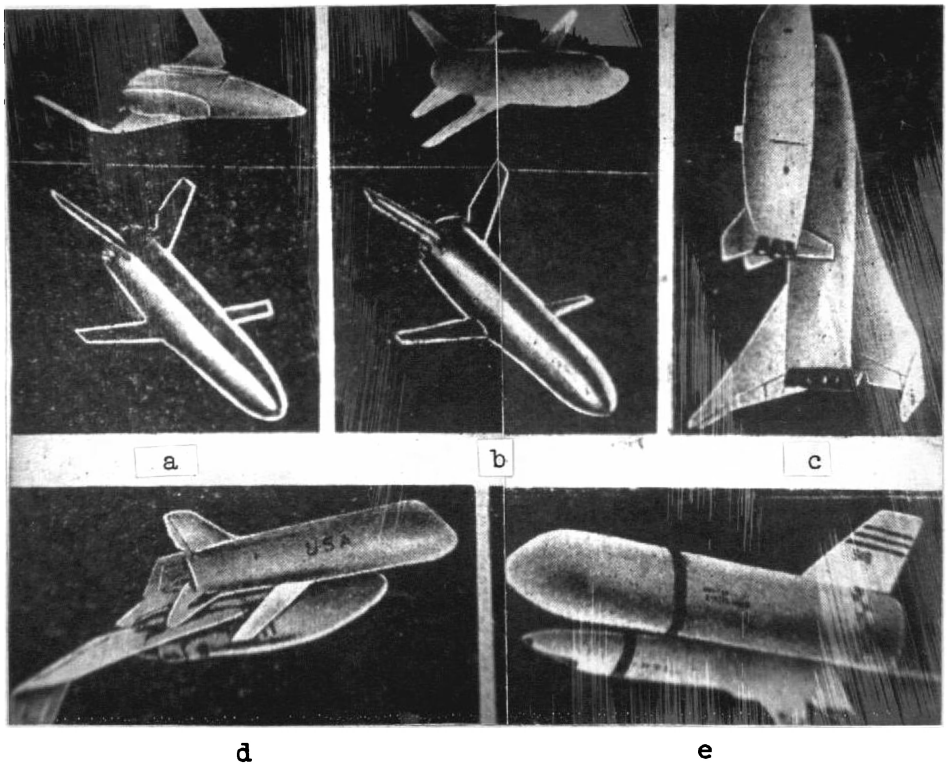


Figure 112. Plans for two-stage transport vehicles.

Figure 112c shows the second stage of a vehicle with a folding wing, and Figure 112d shows the second stage of a vehicle equipped with eleven liquid motors, arranged in three horizontal rows of 2, 4, and 5 motors. In addition to the liquid motors, this stage has eight air breathing motors for flight of the separated stage in the landing region. The straight wing on the second stage allows maneuvers in the lateral direction up to ±300 km. Figure 112e shows the Grumman Company design.

Ground tests are proposed to check the validity of the engineering designs, underlying the choice of vehicle structure and heat shield. The aerodynamic and thermal characteristics, and the stability and control of the transport vehicle, will be determined using wind tunnel models.

Flight tests of the transport vehicle are planned in three phases, in accordance with the technique adopted in development of aircraft. The first phase plans to conduct self-contained tests of the stages at subsonic speeds using air breathing engines, capable of horizontal flight. In this phase, there will possibly also be self-contained tests of the stages in /274 supersonic flight using liquid motors. The second phase proposes tests of the stages using both air breathing and liquid motors. The third phase must begin to test both stages in combination: first suborbital flight, and then flight with injection of the second stage into orbit.

Finally, it can be said that the NASA space vehicle of reusable Earth-orbit class has gone through a complex evolution (Figure 113). According to one of the recent designs, the vehicle should have a payload of 29.5 tons in Earth orbit at height 180 km and inclination 28°, and payload of 11.3 tons at re-entry. The vehicle should be launched vertically, like a rocket, and should land like an aircraft with a landing speed of about 300 km/hour; in the descent from height 45 km to height 15 km, it should have a lateral range of 2000 km. The vehicle should consist of a first stage (two multisection solid fuel units), a jettisonable fuel tank, and a second stage.

The proposed cost of a single launch is to be 10 million dollars. The reusable second stage should make 500 space flights, with a service life of ten years. The layout of the second stage is similar to that of an aircraft using a double delta wing. The dry mass of the second stage is 68 tons. The external heat shield system protects the second stage surface from aerodynamic heating during the injection and atmospheric phases. The basic crew consist of four, and the life support system is designed for a seven-day flight. In addition, the

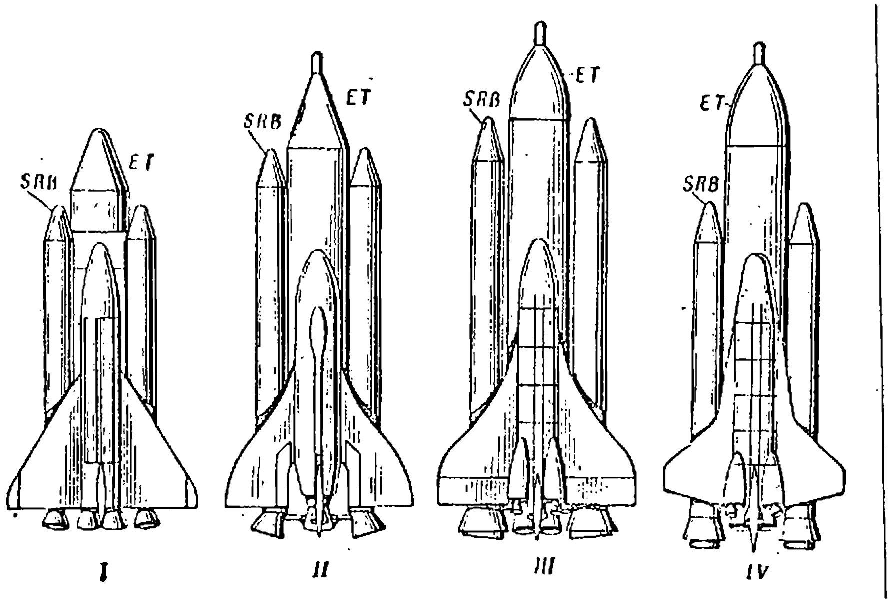


Figure 113. Evolution of design of re-usable space vehicles in the period 1972 — 1973.

I- proposed in March 1972; II- proposed in May 1972; III- proposed in November 1972; IV- proposed in January 1973; SRB is the solid fuel unit; ET is a jettisonable fuel tank.

life support system can provide a thirty-day stay in Earth orbit for the basic crew, and additionally five more persons, at the expense of a reduced payload. The main second stage motor will be three hydrogen oxygen liquid motors, each generating a thrust of 213 tons. The 680 tons of liquid hydrogen and oxygen for the main second stage motor is stored in a jettisonable fuel tank. The main second stage motor should burn parallel to the solid fuel first stage units. In the injection stage, the aerodynamic surfaces of the second stage, apart from the control surfaces, will not be used. At a height of 37.5 km, the solid fuel motors of mass 54 tons will separate from the fuel tank, decelerate, and descend by parachute 360 km from the launch site. The development of the reusable space vehicle is /276
planned in four phases. The first operation of this vehicle is proposed for 1979.

In considering numerous designs for the Earth orbit class of transport vehicle capable of multiple use, we note that, in January, 1973, the American Institute of Aviation and Astronautics published a report in which it made a case for the desirability of constructing a space transport system. It was stated in the conclusions that the success of the USA space program depended on construction of a transport vehicle, that this problem was capable of a technical solution, and did not involve excessive risk. The conclusions mentioned numerous tasks which this system could solve in agriculture, science, and in military applications, and stated that the construction of a re-usable transport vehicle would put an end to discussion as to the advantages of manned or unmanned vehicles. The vehicle would constitute the best combination of manned and unmanned systems, but there are opponents of this system in the USA. One of these is the chairman of the Senate Subcommittee which controls space appropriations, who considers that the building of a transport vehicle at present is not necessary, and that it would be desirable to spend the funds required for this appropriation to build other space vehicles. However, in the USA in 1972, flight tests were made of an automatic system for landing the second stage of the transport vehicle by means of air jet engines. The tests were made with a specially equipped jet aircraft, and, as was mentioned in the foreign press, the tests showed that both automatic and manual landings of the second stage were possible, using existing ground guidance facilities. Graphs were published showing the launch of transport vehicles in the 1978 — 1988 time frame using two launch sites, and a budget was given for the necessary appropriations in the 1975 — 1978 financial years. In the 1974 financial year, it was recommended to appropriate 475 — 500 million dollars for the space shuttle. According to the budget published by NASA, the expenditure to develop transport vehicles in the

1979 — 1990 period is 34,670 million dollars. These vehicles would perform 581 flights in this period (an average of one flight per week). It is also expected that the average payload mass of the shuttle will be approximately 15.4 tons (maximum is 29.5 tons), i.e., the cost for 1 kg of load is approximately 700 dollars, and for comparison, we note that this would be approximately 2000 dollars using launch vehicles. The lateral maneuverability of this kind of system has to be ±2000 km during atmospheric entry, which is also an advantage in comparison with existing systems. The total cost in developing the /277 space shuttle is 5,150 million dollars (at 1971 prices), but, as reported in the American press, this may increase by 30 — 50%.

FLIGHTS TO OTHER PLANETS

The foreign press reports roughly the following concerning a mission to Mars. As regards lunar exploration, the Russians and the Americans are taking different paths. This was shown particularly clearly when the Soviet spacecraft Luna 16 and Luna 20 returned samples of lunar rock to Earth. The lunar vehicles, controlled from Earth, accomplished their successful journies on the Moon. There is no doubt about the validity of the Russian approach to the question of lunar investigation. It can be said with certainty that the Soviet scientists will go on to continued success in the region of planetary investigation using unmanned spacecraft and in the not-too-distant future will try to gather and return to Earth specimens from Mars, which is likely to be the only planet in the Solar System where life is possible.

The United States proposes to begin an intensive search for evidence of the existence of life on Mars. A Mars landing of an unmanned biological laboratory is planned for 1975. This laboratory will conduct a series of experiments which could detect the existence of any Martian microbes, if their biochemistry does not differ too much from that of Earth. The laboratory will also search for amino acids, the basic material of all life on Earth.

The Soviet program and some of the American experiments can be called remotely controlled space investigations.

However, in spite of all the advantages of remote control, it does have some shortcomings. First, it should be noted that scientific observations are extremely complicated. The Soviet and American approach to solution of the problem is fruitful but the search for life on Mars using unmanned vehicles at this stage could end without success, if the living organisms encountered there are rare or if their biochemistry is completely different from that of Earth life. Also, one cannot exclude the possibility that, at present, there is no life on Mars but that it flourished in earlier periods.

The discovery of fossil life remains would be as important as discovering life itself, since it would also confirm the possible existence of life on planets of the other stellar systems. But clearly, unmanned spacecraft cannot find fossils, particularly fossils which are quite different to terrestrial life forms. There may be proposals to equip a spacecraft with television cameras having resolution comparable with that of the human eye. Then we could add wheels and an engine and allow /278
the vehicle to journey over the surface of Mars, on command from Earth. It would search for possible regions where remains of

former Martian life might accumulate, and transmit television pictures to Earth observers for analysis.

A possible program for the operation of a mobile vehicle to investigate Mars could be as follows. First, it will fulfill the function of an observer. Its television cameras will begin to scan the surrounding landscape, transmitting hundreds of pictures to Earth. Then, when the results of observations from closer distance are analyzed, commands will be sent to single out small groups of objects from each section, which in the opinion of the scientists should be studied in detail. The robot will proceed to each such group, its objective being to discover life fossils, and to carry out an analysis of the biological activity. If the results of the investigations seem interesting, a specimen will be taken for subsequent evaluation on Earth.

However, the robot will proceed very slowly over the Martian surface. In the first place, transmission to Earth of a television picture comparable in quality with the picture on the retina of the human eye will take at least twenty minutes, and transmission of a complete panoramic view of the surrounding locality would take about a week. Each series of pictures of potentially interesting sections will require at least a week to transmit, and if an important discovery is made, the transmission of detailed pictures would require still more time. At this rate, investigation of a substantial region around the landing site would take years.

The investigation time will be increased even more because of difficulties in controlling the moving robot at a distance of 56 million miles or more. During automatic investigation of Mars, the robot will probably stop every few meters to

transmit a picture of the sections over which it is traveling. After a delay of several minutes for it to receive the next command, it will move on several meters, etc. However, the rate of investigation can be increased if the mobile vehicle could be taught to analyze what it finds in situ. To do this, one would have to store information in an onboard computer regarding the expected properties of Martian life or its fossils.

Life forms frequently differ from dead material as regards the organized nature of its structure, an example being the annual rings of trees on Earth. Another example is the fossilized remains of water plants, the oldest of Earth life forms observed as yet. It is true that, sometimes, rock crystals can also be evidence of life, if they have a regular structure, but this is much more the exception rather than the rule.

At present, a number of computer programs have been developed which try to find laws even where the relationship is far from clear. The development and introduction of programs of this kind in the proposed space vehicle would not encounter great difficulty. If the robot observed an article with regular geometric shape, it would make a complete logical decision as to whether the item warranted detailed study. The robot should also begin to look for sharp changes in the intensity of light during a survey of close objects using the television eye.

There are programs which allow computers to determine changes in the radiative properties of an object. Such programs could easily be stored in the robot memory before it was sent to Mars. When the robot observed something like life, from the external signs, it would begin to act according to a program stored in its memory.

These are, in brief, some of the proposals published in the foreign press which would facilitate the study of the planet Mars.

Mars is one of the closest planets to Earth. People of old saw Mars, the "red planet", as an ominous sign of imminent disaster. They called it Mars, after the name of the God of War (in Greek mythology). Later, astronomers established that Mars rotates around the Sun at an average distance of 228 million kilometers. The Martian year is almost twice as long as that of the Earth, and is equal to 687 days. The mutual location of the Earth and Mars is constantly changing. Periodically, every two years and 50 days, the Earth and Mars are opposite one another and, on the same side of the Sun. This position of the planets is called opposition. The times when Mars is closest to Earth repeat every 15 or 17 years, and occur in August or September. The orbit of Mars is elliptical. When it is near perihelion, the distance between it and Earth is 56 million kilometers. In size, Mars is about half that of the Earth. The acceleration of gravity on Mars is less by a factor of 2.5 than that on the Earth. Because of the weak gravity, Mars was not able to retain a dense atmosphere during its formation. Like Earth, Mars slowly rotates about its axis. In one day (24.5 hours), it makes a complete revolution. The times of the year on Mars are similar to those on Earth, with the difference that each season lasts twice as long as on Earth. At the hottest place on the planet, and on the Equator, in summer, and at mid-day, the temperature is usually 20°, and in the polar regions, the day air temperature stays somewhat above zero for a period of one-half year. But even in the equatorial region in summer, the night temperature drops to 100° of frost. The reason for this rapid cooling is the low atmospheric density which is not capable of retaining its heat.

At the end of the last century, the remarkable "canals" were first observed on Mars, and then the idea was put forward that there was a highly developed civilization of Martians who had supposedly constructed a complex irrigation system to send water to dry regions.

Radio-astronomical, radar, and infrared investigations have allowed the thermal, electrical, and mechanical properties of the material of the planet Mars to be determined, as well as its thermal conditions. /280

The first space vehicle launched to Mars in 1962 was the Soviet spacecraft Mars 1. In 1965, the American spacecraft Mariner 4 photographed the surface of Mars from a distance of about 12,000 km. In September, 1969, Mariner 6 and Mariner 7 accomplished a flyby of Mars (at a distance of about 3200 km from its surface), and photographed the planet with quite high resolution — 1/3 km. Subsequently, the scientists determined that Mars is not similar to Earth, nor to the Moon. In 1971, the Mariner 9 spacecraft was launched, which continued to study the planet. On May 19, 1971, the unmanned Soviet spacecraft Mars 2 was placed in orbit, which, after flying about 470 million kilometers, went into orbit as an artificial satellite of Mars on November 27, 1971. Its flight continued for 192 days. Capsules were separated from the spacecraft and placed a pennant on the Martian surface carrying the national emblem of the Soviet Union. These three artificial satellites are still in orbit around Mars: the American Mariner 9 spacecraft and the Soviet Mars 2 and Mars 3 spacecraft. The Mars 3 spacecraft sent an entry capsule by parachute into the Martian atmosphere. In July and August of 1973, the Soviet Union launched four unmanned interplanetary spacecraft towards Mars: Mars 4, Mars 5, Mars 6, and Mars 7.

Mars has its own characteristic surface. The spacecraft were able to study the planetary atmosphere by radio methods. Scientific instruments showed that it consists mainly of carbon dioxide. The oxygen impurity is small, and there is practically no water on the planet. It is known that the Earth's atmosphere consists mainly of nitrogen, and it is an adjunct to life on Earth. Not even traces of nitrogen have been observed in the Martian atmosphere, although, of course, this is not confirmation that there are no signs of life there. The atmospheric pressure is very low, about a factor of 100 — 200 less than on Earth.

The successful development of astronomics gives reason to hope that, in the near future, many of the enigmas of Mars will be unravelled. The earliest time for a landing of a manned vehicle on Mars is reckoned to be 1986. The whole operation will take 680 days, of which the astronauts will speed 80 days in the vicinity of Mars. There are several flight plans and variations for the arrangement of this interplanetary mission (Figures 114 and 115). One of these involves the use of the Saturn 5 launch vehicle and a thermochemical motor for the interplanetary spacecraft. Without the propellant and special liquids, this vehicle will have a mass in the range of 28 — 40 /281
tons. It uses hydrogen and liquid oxygen as fuel in all the motors. For flight in one possible trajectory, the required spacecraft launch mass into initial geocentric orbit may be 135 — 180 tons, and the flight duration is about two years. Here the flight speeds are rather high, which makes it difficult to perform a soft landing and, subsequently, to return the probe, which will be launched from onboard the spacecraft, according to the NASA plans. Flights on another trajectory correspond to a flight time of one and one half years, and flights are also possible along trajectories taking one

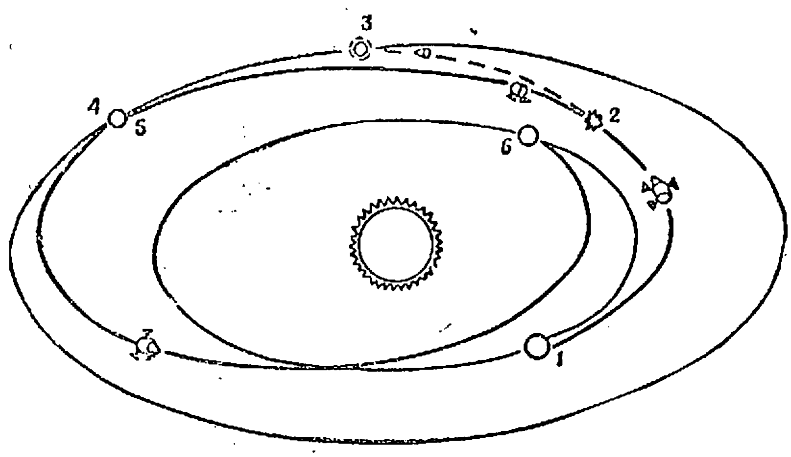


Figure 114. Scheme for planet flyby, with separation of research module.

1- departure from Earth; 2- separation of research module; 3- injection of module into vicinity of planet; 4- planet flyby of main spacecraft; 5- encounter of research module with main spacecraft; 6- return to Earth.

year, with a spacecraft launch mass then exceeding 400 tons. It is mentioned that, at present, the Saturn 5 launch vehicle /282 can put a payload of from 93.5 to 110 tons into low altitude geocentric orbit, depending on the inclination of the orbit. When two stages of this launch vehicle are used with high combustion chamber pressure, the payload can be increased to 145 tons, and when modified motors are used, the payload can be increased to 150 — 173 tons.

It has been determined, from analysis of the mass of several variants of the shuttle system, that it is best to divide the spacecraft into two main units: the bus and the interplanetary unit. The bus will constitute the payload for a second Saturn 5 launch vehicle. The entire spacecraft will be sent into an initial geocentric orbit in the form of two payloads on two launch vehicles, and will subsequently rendezvous and dock. Figure 116 shows a spacecraft designed for an active flyby of Mars and separation of the research module. The same

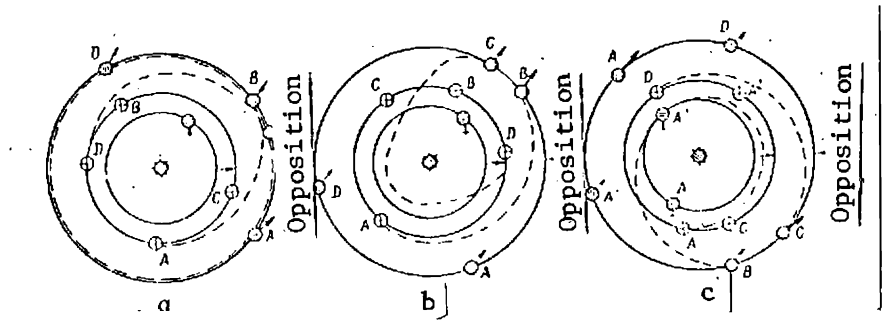


Figure 115. Possible arrangements for Mars mission.

a- flight with long period near Mars; b- flight with short period at Mars; c- flight with perturbation maneuver at Venus on the Earth-Mars leg; A- situation at launch of spacecraft from geocentric orbit; A'- situation at perturbation maneuver near Venus; B- situation at arrival of spacecraft at Mars; C- situation at departure of vehicle from Mars; D- situation at return of vehicle to Earth.

picture shows the bus, attached to the spacecraft, which will be made ready prior to launch. The command and service modules can be moved to the central part of the structure during the launch, which will allow the effect of loads on the joints to be decreased during orbit injection. During injection of the spacecraft into its initial orbit, the crew will be in the command module, which is located immediately attached to the emergency rescue system immediately before the launch. The joints are folded and covered by the main shroud. After injection, the shroud is jettisoned and the joints are unfolded into their required position. After checking the spacecraft systems, the spacecraft will rendezvous with the capsule, for which attitude control is controlled from the spacecraft. Then mooring and docking of the spacecraft to the capsule is accomplished automatically, and, thereafter, the spacecraft is ready for launch.

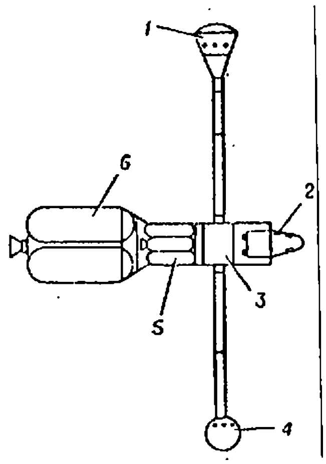


Figure 116. Scheme for basic spacecraft.

1- command module (entry capsule); 2- research module; 3- shelter for crew in the event of solar flares; 4- service module; 5- motor for active flyby; 6- motor for departure from Earth.

Later, it was explained that the principle of a flyby of a planet and separation of a research capsule will allow a considerable reduction in the spacecraft launch mass, since the operations near the planet are then performed by a spacecraft of minimum mass, while the main spacecraft bus continues its motion in the flyby trajectory. The use of this principle, in conjunction with elliptical parking orbits relative to the target planet enables a considerable reduction in the launch mass of the interplanetary spacecraft intended for flight to Mars; the spacecraft is injected into an initial geocentric orbit using existing or modified Saturn 5 launch vehicles.

The spacecraft, consisting of two independent units, the acceleration unit for departure from Earth, and an independent interplanetary spacecraft, has an optimum construction. Each of its units can be injected, as has been mentioned, into an initial orbit by an independent launch vehicle. After rendezvous and docking, these units form the complete spacecraft, and this does not require complex operations, associated with orbital welding and checking of a large number of equipment elements. It is considered that it will be entirely possible to carry out interplanetary missions in future years, using present propellants, launch vehicle motors, and technology. /283

It seems logical that, to send missions to parking orbit in the vicinity of a planet requires a spacecraft launch with a larger launch mass than for a flyby mission. This leads to the idea that thermochemical motors are inappropriate for such spacecraft. It is suggested that nuclear motors will be used for this case. For a parking orbit around the planet for ten days and when a spacecraft is launched into circular parking orbits relative to the planet, the required spacecraft launch mass for a flight to Mars is from 900 to 2700 tons. Thus, in this case, in addition to a clear need to use nuclear motors because of the large launch mass, there are also difficulties associated with planning a space program for future years, since the spread in the required spacecraft launch masses for flights to Mars is very great. However, if the spacecraft is injected into an elliptical parking orbit relative to the target planet, a marked decrease results in the required characteristic velocity or in the launch mass.

As has been mentioned, a decrease in the spacecraft launch mass is possible if one uses a flyby of the planet and separation of the research module; for this scheme, the main trajectory (i.e., the trajectory of the main spacecraft) is a flyby trajectory with return to Earth. Before the spacecraft reaches the planet, the research module separates and enters a planetocentric orbit to conduct its scientific investigations. After a given interval of time spent near the planet, the main spacecraft bus flies on, following a standard flyby orbit. The research module descends from parking orbit and accomplishes a rendezvous at hyperbolic speed (relative to the target planet) with the spacecraft bus, and returns to Earth. Since only the /284 research module maneuvers near the planet, the total spacecraft launch mass into the initial geocentric orbit is considerably reduced.

Use of planetary flyby schemes and separation of a research module, in conjunction with an active flyby, allows reduction of the launch mass for flights to Mars to 450 tons in unfavorable launch years, and to 240 tons in favorable years. Therefore, two Saturn 5 launch vehicles (particularly of the appropriate modification) are sufficient to inject spacecraft into initial orbit for some of the missions to Mars.

The USA is developing a special Viking system, to land a spacecraft on the surface of Mars, to investigate possible life forms there. The flight to Mars will take 305 — 360 days.

The spacecraft will consist of an orbiter and a lander, which will be placed in orbit around Mars together. Flight trajectory correction will be made using the orbiter motor. The orbiter also provides the spacecraft electrical supply (solar cells). During the period in orbit, the spacecraft will use two television cameras to investigate the Martian surface and will use an infrared spectrometer and a radiometer to obtain data on the temperature and water vapor content. This information will be transmitted to Earth, where it will be used to construct maps of the Martian surface. The orbiter will also carry equipment to investigate the atmospheric structure of the planet and equipment for trajectory measurement.

The minimum lifetime of the orbiter following injection into orbit around Mars is proposed as 140 days. The information obtained from the orbiter must be used in part for reappraisal of the proposed landing areas and choice of new areas. Before separation from the orbiter, the lander will jettison its shroud, which has ensured its biological isolation. The lander will begin its descent from a height of 244,000 m. At an altitude of 6100 m, the heat shield will be jettisoned and the braking

parachute will open. At a height of 1586 m, the retromotor will begin to function. The spacecraft attitude will be controlled by an inertial navigation system, which will operate in conjunction with a wide band radar altimeter and a multipath Doppler radar. It is supposed that Martian winds may complicate the landing, and, therefore, at a height of 3 m from the surface, the retromotors will be switched off. From this height, the lander will fall freely and make a soft landing at the landing site. During the descent, the mass spectrometer will measure the composition and structure of the atmosphere. At the same time, measurements will be made of pressure, temperature, and atmospheric density. The data from these measurements must be transmitted to the orbiter, whence they will be retransmitted to Earth. After the landing on the surface of the planet, /285
communication with Earth will be established either via the orbiter using the UHF antenna system, or directly via the lander, operating for a period of 2 — 4 hours per day. One important part of the lander is the computer, which will be used to control the landing on the planetary surface, and also to monitor the operation of all the lander apparatus.

During its 90-day period on the surface, the lander will conduct biological, molecular, optical, meteorological, and seismological investigations, and will also study the magnetic and physical properties of the planet.

In March, 1972, the USA launched the unmanned Pioneer 10 spacecraft, designed to investigate Jupiter on a flyby trajectory. The length of the flight path to Jupiter is roughly one billion kilometers. The launch vehicle gave the spacecraft a design speed of 13.8 km/sec, the highest speed for any spacecraft launched from Earth at present. For electrical supplies, radio isotopes are used. Fifteen minutes after launch, the

spacecraft separated from the last stage of the launch vehicle. It is spin stabilized. According to calculations, in July 1972, the Pioneer 10 passed into the asteroid belt between the orbits of Mars and Jupiter. The thickness of this belt is 280 million kilometers. On December 2, 1973, the spacecraft passed at its minimum distance from Jupiter, about 14,000 km. At 11 — 12 years from launch, the spacecraft will pass out beyond the Solar System. It is hoped to continue communication with the spacecraft for 5 — 6 years, until it is at a distance of 2.4 billion kilometers from Earth, i.e., it is between the orbits of Saturn and Uranus.

The Pioneer 10 carries a telescope counter, designed to study radiation in the upper layers of the Jovian magnetosphere. The flight will give information on the extent of the Jovian radiation belts, and information concerning their differences from the radiation belts of the Earth. Because the spacecraft, in flying beyond the Solar System, could encounter extraterrestrial civilizations, it carries an aluminum plate with a picture giving a representation of the civilizations of those who launched it, and also concerning their location in the Universe, i.e., concerning the Solar System. The plate has pictures of a man and a woman, and the man has his arm raised in a gesture of greeting. There is a picture of the hydrogen atom, and also the Sun amidst some pulsars in the position occupied at the moment of launch of Pioneer 10.

According to reports and the foreign press, the NASA /286
management are preparing a general plan for space developments. It includes a reflection of scientific investigation and engineering problems which will be solved in the interplanetary space near the Moon, and also in the vicinity of the Earth and planets using manned vehicles. The development of orbital spacecraft

and orbiting aircraft will be in keeping with all the operations in which it is proposed that man take part.

A preliminary graph showing operations in the field of astronautics is given in Figure 117. It can be seen that the first three fields deal with study of the Moon (the Apollo program). The "grand tour" program, which envisioned launch of unmanned spacecraft for flyby missions to the outer planets of the Solar System — Jupiter, Saturn, Uranus, Neptune, and Pluto — was planned for the 1976 — 1979 period. Then it was decided to replace it by launch in 1977 of two unmanned spacecraft of the Mariner type for a flyby of Jupiter and Saturn. The flight to Jupiter would take about a year and a half.

A great deal of attention is given to construction of orbital laboratories in different orbits, including a laboratory for near planetary operations, and it is also intended to launch manned interplanetary spacecraft. The graph shows that there is no decrease in efforts to create experimental hypersonic vehicles with large lift/drag ratio, which is intended to create a shuttle aircraft for transportation between geocentric and selenocentric orbits. The first orbital flight of the space-atmosphere shuttle vehicle (Figure 118) is planned in the USA for 1978. /287

To solve these tasks, work will be performed to create energy sources specialized for long duration space flight, including the construction of a nuclear engine and installations with thermoelectric conversion system.

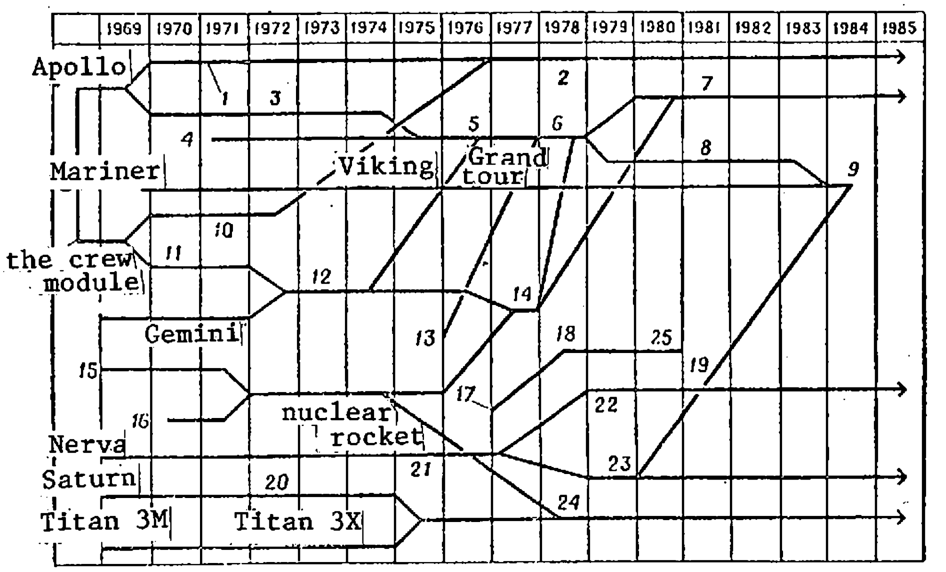


Figure 117. Tentative graph showing USA space development over the next fifteen years.

1- investigation of the Moon; 2- orbital laboratory as a space bus in selenocentric orbit; 3- cessation of orbital investigations with the Apollo system; 4- development of space laboratory; 5- test specimen laboratory; 6- creation of laboratories in orbit; 7- orbiting space base; 8- modified space laboratory for near planetary operations; 9- launch of manned interplanetary spacecraft; 10- modified lunar spacecraft; 11- modified basic Apollo unit; 12- orbiting aircraft, developed from existing spacecraft; 13- retransmitting artificial Earth satellites; 14- future orbiting aircraft; 15- development of experimental hypersonic vehicles, HL-10, X-24, etc.; 16- development of future orbital laboratories; 17- the SNAP power facility with thermoelectric conversion system; 18- SNAP using a heat cycle; 19- future energy facility; 20- intermediate variants of the Saturn launch vehicle; 21- nuclear rocket motor; 22- orbiter aircraft for transportation between geocentric and selenocentric orbits; 23- rocket stage for interplanetary maneuvers; 24- launch vehicles (low cost first stages); 25- self-regenerating system.

It is clear that experience will introduce corrections in the graph of space operations in the next fifteen years. Future plans for space operations are being developed not only in the USA, but also in other countries.

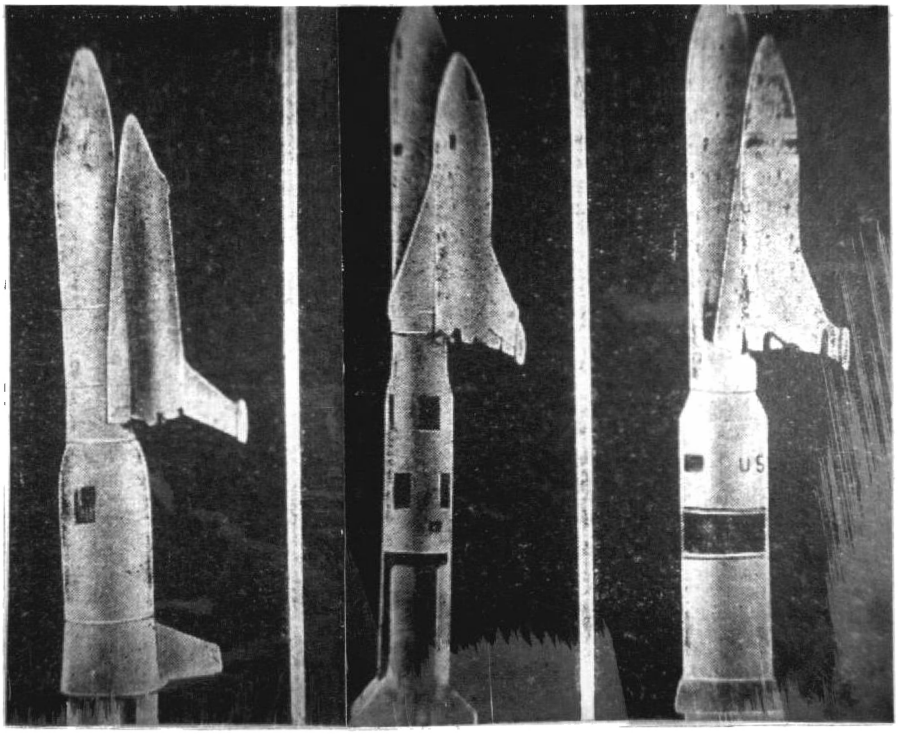


Figure 118. Concepts in the development of a two-stage shuttle vehicle.

The success of astronautics in the thirteen years following the space flight of the citizen of the Soviet Union Yurii Gagarin has confirmed that the conquest of space by man is going forward at a present tempo which it would have been difficult to predict 10 — 15 years ago.

CONCLUSIONS

The extraordinary scope and rate of accumulation of diverse scientific information is leading to an accelerated discovery of nature's secrets by scientists, to a rapid evolution of scientific theories, and to conceptions of the world as an entity.

Astronautics is giving science new and more powerful methods of investigation. It is clear that there is no scientific field which has not taken part in the conquest of space, and has not in turn grown in scope and depth. The conquest of space has allowed us to carry scientific instruments and humans beyond the limits of the Earth, to investigate the microprocesses and macroprocesses occurring in space. The progress of contemporary and theoretical mechanics and cybernetics, physics and chemistry, is closely associated with the development of space technology. The construction of space rockets has required intense development in the mechanics of continuous media, the mechanics of variable mass, and the theory of stability. The improvement of systems for control of space vehicles has required very great development in the theory of optimum regulation and automatic control.

The group of basic sciences, which study nature and man in the context of the need to give life support in space conditions, requires solution of new problems in the areas of biochemistry and biophysics, general physiology, physiology of the higher nervous activity, general psychology, and emotional psychology. The achievements of science, obtained in space technology, have

assisted very greatly in the solution of important practical problems. Astronautics has promoted multifaceted investigations of the Sun, our huge natural thermonuclear reactor, and this has drawn us closer to the age of thermonuclear energy on Earth. The space investigations of the Moon have made it possible to obtain geochemical, mineralogical, and geological data on our own planet. By the use of astronautics, we are able to understand more precisely the mechanisms involved for formation of useful minerals, both in the surface layer, and in the entire thickness of the Earth, and to determine the location and the amount of underground resources. There has been improvement in the quality of prediction of various changes in the Earth's core, and this makes an impact on architecture, agriculture, transportation, and communication.

The use of satellites and orbiting spacecraft only in agriculture, geology, hydrology, oceanography, geography, meteorology, television and radio communication produces great savings. Therefore, in the recommendations of the XXIV Congress of the Communist Party of the Soviet Union at the Ninth Five-Year Meeting, tasks were assigned for conduct of scientific work in space in order to develop telegraphic and telephonic communication, television, meteorological investigations, and to accomplish other technical tasks using manned and unmanned spacecraft.

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